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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

7 RUE ANCELLE 92200 NEUILLY SUR SEINE FRANCE

AGARD CONFERENCE PROCEEDINGS 522

TacSats for Surveillance, Verification and C3I

(Les Satellites Tactiques (TacSats)
pour la Surveillance, la Vérification et la C3I)

Papers presented at the Avionics Panel Symposium held in Brussels, Belgium from 19th—22nd October 1992.

NORTH ATLANTIC TREATY ORGANIZATION

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North Atlantic Treaty Organization

Organisation du Traité de l'Atlantique Nord

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Theme

The recent Desert Storm operations underscored the essential need for space systems to fight and win a theatre war. An emerging concept for space systems currently receiving widespread attention for the 90s is characterized by the term "TacSats" (also called Cheap Sats, Light Sats, Small Sats etc.). These satellite systems are characterized by relatively low costs and lightweight satellites with performance capabilities focused at meeting theatre command requirements. This is in contrast with complex, heavy satellites focused on meeting national requirements. The TacSats concept may be particularly appropriate for the current environment of declining defence budgets and conventional force reductions. These systems offer capabilities which can be purchased incrementally in contrast to traditional space systems which require future expenditure. They can be launched by smaller, less expensive launch vehicles, perhaps controlled by the local theatre commander.

TacSats may offer a flexible economical solution for NATO's surveillance, communications, and command and control needs in the 90s. In an environment of reduced forces, it becomes more important to know precisely where potential enemy forces are located, what is their strength, where they are going, etc. This is vital in order to operate effectively with smaller forces. Space systems offer an effective and perhaps the only means of performing these functions. Affordable TacSats developed for NATO may offer a way to counter a changing threat in an environment of reduced budgets and force structures.

The Symposium addressed the following topics:

TacSats Systems
- NATO architecture and system engineering
- Payload concepts and designs
- Spacecraft bus concept and designs
- Launch vehicles for the SATS
- Mission control and user equipment

Space System Technologies of the Future
- EHF, SHF, UHF antennas, receivers
- Optics and electro optics components
- IR detectors, focal planes, coolers
- Materials and structures
- Communication and telemetry
- Data processing, on-board processing
- Power systems, solar cells, batteries
- Built-in test and non destructive testing
- Propulsion technologies.

Thème

Les opérations récentes «Tempête du Désert» ont souligné le besoin vital de se doter de systèmes spatiaux susceptibles de mener et de gagner une guerre de théâtre. Un nouveau concept de systèmes spatiaux pour les années 1990, qui suscite beaucoup d'intérêt à l'heure actuelle, est caractérisé par le terme «TacSats» (appelé aussi CheapSats, LightSats, SmallSats etc.). Ces systèmes sont caractérisés par des coûts relativement modiques et des satellites légers avec des performances étudiées pour répondre aux besoins du commandement du théâtre, par opposition aux satellites lourds et complexes destinés à satisfaire aux besoins nationaux.

Le concept «TacSats» paraît particulièrement adapté au contexte actuel de compression des budgets de défense nationale et de réduction des forces armées. Les capacités offertes par ces systèmes peuvent être acquises de façon progressive, contrairement aux systèmes spatiaux classiques, qui sont toujours accompagnés de coûts additionnels. En outre, ils peuvent être mis sur orbite par des véhicules de lancement plus petits et moins coûteux, commandés éventuellement par le commandant du théâtre local.

Les TacSats semblent offrir une solution économique et souple au problème des besoins de l'OTAN en matériel de surveillance, de télécommunications et de commandement et contrôle pour les années 90. Dans une situation de forces réduites, il importe de savoir où les forces ennemies potentielles pourraient être situées, quel est leur nombre, quelle est leur route etc. Ces informations sont indispensables à tout déploiement efficace d'un nombre réduit de troupes. Les systèmes spatiaux offrent peut-être la seule possibilité d'exécuter ces fonctions d'une manière efficace. Dans ce cas, des TacSats développés pour l'OTAN, à un prix abordable, permettraient de contrer une menace qui évolue, dans un monde où les budgets et les structures des forces sont en diminution.

Le symposium a traité des sujets suivants:

Systèmes «TacSats»
- Architecture et ingéniérie des systèmes OTAN
- Concepts et études de la charge utile
- Concepts et études des bus pour véhicules spatiaux
- Lanceurs SATS

Technologies de demain pour systèmes spatiaux
- Antennes et récepteurs EHF, SHF et UHF
- Composants optiques et électro-optiques
- Détecteurs IR, plans focaux, refroidisseurs
- Structures et matériaux
- Télécommunications et télémétrie
- Traitement des données et ordinateurs embarqués
- Réseaux d'alimentation, cellules solaires, batteries
- Dispositifs de test intégrés et essais non destructifs.

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TECHNICAL EVALUATION REPORT

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SUMMARY

The symposium was highly successful in that it brought together a broad spectrum of the NATO space technical community with quality papers on the missions, applications, and technical aspects of TACSATs. The main topics centered on surveillance and communication systems, and the issue of determining requirements. At the completion of the symposium, it was clear that state-of-the-art total systems could be built that would be responsive to the theater commander at a reasonable cost. While the level of technical detail varied; that is, more emphasis on the satellite options than the ground systems, there was sufficient detail to make the case for the validity of the TACSAT concept as discussed in the symposium theme.

The output from this symposium will be the central point of departure for the Working Group 16 efforts on TACSATs for NATO. In this regard, the symposium was an outstanding success.

TECHNICAL CONTENT

The symposium was opened with an address by Robert Dickman, Brigadier General, USAF, who is the Deputy Chief of Staff for Plans at Air Force Space Command in Colorado Springs, Colorado, USA. His perspective was from the warfighter's point-of-view within which he challenged the audience of mostly scientists and engineers to focus on the needs of the user; emphasizing, that if TACSATs are to be a part of space in the future, they must be measured in terms of th.ir unique contributions to mission success.

To make his point, Gen. Dickman expressed the belief that space will be a dominate factor in any future conflict; and, as a result, will have to improve, not decrease, in the future. He emphasized that while this belief is widely accepted, in an age of decreasing resources, requirements and affordability must come together. Therefore, the developer must seek out the view of the warfighter in an effort to identify critical space deficiencies. In this regard, the developer will be dealing with a customer who is smarter, who will demand more from space, and who will insist that the space capabilities be there during hostilities.

To emphasize these points, Gen. Dickman gave a space "report card" on Desert Storm.

> Communication - never enough, not mobile enough, and the terminals were inadequate.
>
> Weather - good information, but the data dissemination was poor.
>
> Navigation - great, but not enough receivers.
>
> Imagery - a controversial subject. While there was not enough surveillance, it was actually viewed as a failure because of dissemination problems.

As a result, the overall space rating can be viewed as follows:

> Capability - great
> Payload tasking - moderate
> Dissemination - terrible

He went on to emphasize that the reason for this "card" rating goes back to what the warfighter wants. He wants: continuous availability, flexibility and availability to the end use.. Put another way, from a combat perspective the warfighter will ask:

- Can I get what I want when I want it?
- Can I distribute it to my forces?

In closing, Gen. Dickman challenged the symposium to focus on the deficiencies.

1. Improve distribution
2. Improve data streams.
3. Develop new space systems; but, avoid uniqueness, concentrate on interoperability, and ensure that we can exercise as we fight.

The challenge for TACSAT will be:

- What are the needs?
- What are the options?
- What is possible now?
- What are the "measures of effectiveness?"

Session I - TACSAT Concept and Need

The intent here, was to "set the stage" - WHAT ARE WE TALKING ABOUT?

In this session, as with almost the entire symposium, two mission areas received the most attention - surveillance (imagery) and communications. Of those two mission areas, surveillance took the "lion's share" of the group's attention, with most of the discussions centered on resolution.

From the four papers, there were several areas of consensus, two areas of divergent opinions, and one outstanding issue.

Consensus

- Total system designs must provide for flexibility, responsiveness, on-demand support, and high revisit times,

- That TACSATs can be used in tandem with core systems by: raising mission area responsiveness, selecting orbits to match the threat, concentrating on a specific geographic area, and filling in for a failed primary system.

- They must generate a field useable product (surveillance orientation).

- Any new design must be compatible with current terminals and allow for realistic training.

- Affordability must be paramount through commonality, a common bus approach, using items off-the-shelf, and controlling requirements.

- While it was generally agreed that TACSATs fall in the weight regime' of 300-700 kg, the validity of continued use of this definition became an area of divergent opinion.

Divergent Opinions

Within this session, and as the symposium progressed, there was continued discussion concerning two opinions. While there was no consensus on these opinions, it was agreed that Working Group 16 must look at them further. They are:

1. Large systems can be considered as TACSATs if data dissemination and payload tasking can be more responsive to the warfighter.

2. TACSATs do not necessarily have to be launch-on-demand systems; they can also be stored on orbit in a dormant state ready for rapid activation and repositioning.

Issue

There was one issue - how do we come to grips with a formal statement of the need. Imbedded in this issue is whether a statement of need is necessary for TACSAT or if TACSAT is a technical option whose "time has come" and it is now ready for consideration in relation to a broader sphere of needs.

Session II - TACSATs - Aspects

This session covered a broad range of TACSAT concerns ranging over the areas of data collection, system control functions, viability of a multimission bus, and a model for assessing various COMSAT options.

While not giving specific answers to TACSAT command and control, the session stressed the importance of this area and how such answers might be addressed in future analyses. In the "bus" area, the point was made that such bus's have been flown in an R&D mode and that reasonable bus's could be provided for TACSAT low earth orbit missions.

Session III - TACSAT Applications - Systems

A wide variety of mission areas was discussed; however, the greatest emphasis was in navigation, communications and surveillance. In general, the technical emphasis was on techniques for moderate capable systems with upper limits on spacecraft weight and reduced complexity. Of particular note, was the overwhelming attention to limiting the warfighter support to specific geographic areas rather than global or continental support. These views can be seen in the following distillation of the output from four papers presented in this session.

Navigation

A concept was discussed that used four to nine satellites at geostationary orbit versus the medium earth orbit of GPS. In keeping with system simplicity, lower cost (through lighter weight), and limited geographic coverage, the concept had the following features:

- Equivalent GPS capability

- Ground based atomic clock for lighter weight and simpler spacecraft

- Coverage limited to Europe, the Mid-east, and Africa

- Three satellites launched simultaneously on Ariane.

Communications

The discussion here centered mostly on comsat applications as they relate to Air Command & Control. Of particular note was the point that TACSATs are needed in all areas and that their configurations should include:

- Bi-directional
- Broadcasting
- Ground to ground
- Ground to air

However, communication type TACSATs were seen as a primary means for alleviating the problem of Air Tasking Order dissemination.

While the discussions in this area presented new thinking in how COMSATs could contribute to the air campaign and, to some degree, ballistic missile defense; more analysis is needed to determine the value of TACSATs as a specific class of satellite. This is consistent with the issue raised in Session I concerning the use of large satellites as TACSATs.

Surveillance

This area received the most attention in this session, appearing in three out of the four papers.

As with communications, surveillance satellites were presented as key elements in developing the Air Tasking Order and in supporting ballistic missile defense. The function of surveillance satellites was considered prime in detecting airbase activation, raid detection, launcher detection, detection of missile launch preparation, and missile launch detection. However, specific requirements that allow for the use of TACSAT class satellites have yet to be defined.

Some time was spent on the potential roles TACSATs might play from a systems point-of-view. It was emphasized that surveillance TACSATs could:

- Reduce system complexity
- Enhance observation capabilities (more from quantity than quality)
- Achieve high revisit of specified sites
- Provide a quick concentration of resources.

Here again are the points that TACSATs play best in activities that are geographically localized and that require quick reaction to a situation.

At this point, came the first discussions of the technical and operational viability of certain levels of resolution, orbital altitudes, spacecraft weight, and surveillance techniques.

- Resolutions seemed to fall in the 3-5 meter regions, with some excursions to 1.4 meters.
- The orbital altitudes ranged from 280 km to 600 km.
- The spacecraft weights were in the 600 to 750 kg range.
- Surveillance techniques included electro-optical and synthetic aperture radar (SAR).

At issue in this area was the acceptable resolution. As always, the user asks for the best; however, it was generally agreed that best in surveillance means large, expensive, and limited in quantity. This item remained as a contentious issue throughout the symposium.

Session IV - Panel Discussion

The objective of this session was to engage the audience in addressing any issues and concerns that may have come up in the prior sessions. Four major issues were discussed, although no resolutions were agreed upon. The issues were:

- How are requirements to be established?
- Should TACSAT be strictly theater oriented?
- Will TACSAT be complimentary to the major space systems?
- What launch strategy is appropriate?

These issues will be a prime concern of Working Group 16.

The value of this session was not in the discussions, but in the bringing forth of these four key issues.

Session V - Communications Concepts

In this session the emphasis was on technology and how it might lead to TACSAT type communication satellites. The main emphasis was on EHF; although there was some discussion of SHF.

The flow of the session was particularly well done with Dr. Ince leading off with a discussion of Working Group 13's efforts on NATO Satellite Communications, which was started in 1986. In the discussion, he reminded the audience that our past emphasis was oriented to the strategic mission where the systems had to be highly survivable, ensure continuity of service, provide ECCM, and provide low capacity communication for emergencies.

But times have changed. Dr. Ince emphasized that satellites should be smaller and cheaper, with launch flexibility by employing systems such as clustered satellites. He also emphasized some movement from SHF to EHF. Most importantly, Dr. Ince provided a listing of NATO actions —

- Define NATO and National requirements
- Develop and agree on an architecture
- Define the technologies
- Encourage companies to join together

The remaining papers discussed the technologies that will make smaller EHF satellites possible and some of the problems that might be expected at these frequencies. In general, the emphasis was to provide technologies that would allow more users to take advantage of the AJ characteristics of EHF while operating at medium data rates rather than low data rates. The technology emphasis was on variable beamwidth antennae, frequency synthesizers, and processors. In the case of the antenna technology, the objectives were for easier operation in elliptical orbits and for ease of switching geographic locations for satellites in geostationary orbits. For frequency synthesizers and for processors, the emphasis was on significant weight and power reductions that would allow for payloads in the 100 kg and 290 w regime's with terminal capacities of 27 LDR channels at 45 kbits and 17 MDR channels at 3 mbits.

Other discussions included MDR EHF synchronization techniques for minimizing acquisition times and an examination of interference from mountains and foliage.

Essentially, this session verified that the technology is available for TACSAT type communication satellites, particularly at EHF. The question remained concerning the adequacy of such systems for military needs. Certainly, the capacity of such systems can now be defined with great confidence and no doubt the same can be said for cost. The question is, is the terminal capacity described above cost effective under a range of scenarios?

Session VI— Launch Systems

This session confirmed that a broad range of launch options will be available after 1995 to support TACSAT type payloads as identified in Session I. These boosters, in increasing weight carrying capability, are: Pegasus, Taurus, Delta II, MLV-3, Atlas II and Ariane.

The question that remains to be addressed is what deployment strategy will be used for TACSAT. If the booster is the limiting factor, then strategies are limited to the following:

Booster	Store-on-Orbit	Rapid Launch
Pegasus (1)	X	X
Taurus (2)	X	X
Delta II	X	
MLV-3	X	
Atlas II	X	
Arainc	X	

(1) Max 500 kg LEO
(2) Max 1500 kg LEO
Max 500 kg GTO

Therefore, smaller TACSATs can have multiple options where as the large TACSATs (particularly to GEO and elliptical) must be stored on orbit or settle for 20 to 60 days turn-around.

It was generally agreed that it would be prohibitively expensive to convert MLV-3, Delta II, Atlas II, and Ariane to turn-around times that are commensurate with Pegasus and Taurus. On the other hand, the MLV-3, Delta II, Atlas II, and Ariane will be cost competitive for store-on-orbit because of their ability to launch more than one TACSAT; the issue being, will the military situation at a given instant allow for such an approach? Working Group 16 will have to address this issue.

Session VII – Spacecraft Bus

The question of building a common bus with the appropriate attitude control was addressed in this session. In general, the papers confirmed that low cost common spacecraft buses can be built, although it was also recognized that some penalties would be realized from over optimized or sub optimized subsystems.

The one paper that specifically addressed the bus design pointed out that for spacecraft with single function oriented payloads, the entire system (ground system, payload, spacecraft) could be developed in 12-15 months for $15-20M. Where as more complicated systems could cost $50M. This later case was the US Navy GEOSAT Follow-on. It was emphasized during the question and answer period, that in the case of the 12-15 month example, there was no redundancy and the design life was one year.

The above discussion does not completely make the case for a common TACSAT bus. However, the point is made that not every space project must be expensive.

The Working Group 16 may have to further address the common TACSAT bus approach.

The third paper in this session looked at electric propulsion uses for TACSATs in the weight ranges of 300 to 800 kg. The two uses of electric propulsion were for orbit keeping tasks and for orbit raising (from 400 km to 1400 km). It was concluded that:

- Ion propulsion was the preferred technology for orbit keeping.
    - Significant mass savings
    - However, long lifetime ion thrusters have yet to be demonstrated on orbit

- Low power arcjets are the best candidates for orbit transfer and raising.

In general, more work is needed for electric propulsion; however, it should be supported and examined for introduction into any TACSAT program.

Session VIII — Advanced Technology

This session was divided into two discussions – the DARPA (USA) space technology efforts and CAMEO.

The DARPA technology discussion told of significant strides being made to reduce the cost of doing business in space. It is more appropriate for the reader to examine the paper than to present a detailed summary here. However, a brief summary is appropriate.
This program includes:

- Pegasus
- Taurus
- DARPASAT
- Optical technology for light weight systems
- Submarine laser communications technology
- EHF technology
- Satellite subsystems
- Common buses

An example of the progress being made can be seen in the EHF technology work. This technology is on its way to lowering spacecraft weight and power by >50%.

The overall program will use smaller satellites to quickly validate technologies so large satellites can be procured using more advanced technology.

A description of CAMEO was presented. Its objectives are:

- Multi-spectral small satellite
- DOD/civil environmental and weather
- Direct downlinking of data
- Use of a common bus

During the question and answer period, the status of CAMEO was requested. The answer was that funding was deferred by Congress even though it had the full support of DOD.

Session IX — Radar Concepts

As with the following session, this session addressed a specific technique for battlefield surveillance - using radar in the synthetic operture mode. The papers were in-keeping with the general approach of TACSAT, that of supporting theater operations. This point was shown as pivotal in allowing a reasonable weight spacecraft so that it fit the smaller class satellite category. In general, the spacecraft weights fell in the 500 to 800 kg regimes. In the past, most radar spacecraft weights were in the 5000 kg class; so why the difference? Basically, the difference was in the size of the region of concern (2000 x 2000 km), which in turn reduced the duty cycle (5-30%) which reduced the weight and, to some extent, reduced the size of the antenna. Other spacecraft parameters fell into the following areas:

X-band or C-band
300 to 500 km altitude
200 to 400 mbit data rate
6 to 8 satellite constellation
Phased array antenna
4 x 2 meter antenna
2-5 meter resolution
30 images per pass

There was a unanimous call for a demonstration flight. The overall feeling was that a SAR spacecraft, built to the above specifications, is state-of-the-art.

Additional papers were presented that discussed lightweight, store and forward microwave applications, a maritime application of a SLAR and an altimeter to determine wave heights.

Session X — Electro-Optic Concepts

This discussion followed the pattern seen in the SAR session. In general it was believed that state-of-the-art E-O TACSATs could be built according to the following:

1-5M resolutions
250-450 km altitude
350 to 650 kg spacecraft weight
200 to 300 mbs data rate

Three additional points were made: (1) the ground station could be air transportable; (2) a convincing demonstration could be conducted for $50-100M; and (3) a $50M price per satellite is achievable.

As with the radar SAR, the group felt a demonstration was absolutely necessary to get something into the hands of the user.

Session XI — Panel Discussion

This session reviewed the outputs from each session followed by questions and answers with the audience. The discussions centered on three themes — can we achieve the TACSAT objectives, how much resolution can we expect from surveillance systems, and how do we identify requirements.

RECOMMENDATION:

From the content of the papers, it appears that TACSAT type spacecraft for surveillance and communications could be of great value to NATO. While such systems will be more affordable than larger more capable systems, they will not be procured by any one member nation and therefore should be approached as a joint international effort.

These are three specific recommendations: (1) These papers should be used as the foundation for the Working Group 16 efforts. (2) The results of Working Group 16 should then be briefed to the member nations and NATO headquarters. (3) A NATO team should be established to develop technical and operational options, with costs, to be reported out to NATO by early 1994.

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ABSTRACT

The concept of a Tactical Space System, "TACSAT" is a means to provide a rapid, on demand, augmentation of the backbone U.S. military space systems. Such augmentation would be valuable to temporarily replace lost capability or in times of crisis, to accommodate surge demands. Because augmentation needs are not always known a-priori, it would be desirable to be able to rapidly constitute the appropriate payload-satellite bus combination to accommodate the need for a specific space capability. To do this, one can envision a standard bus capable of accepting a variety of payloads, or better yet, a single spacecraft designed to perform several different missions. Both options are considered. A number of potential missions exist in the areas of surveillance, navigation, environmental sensing and communications. Of these, two are presented as strawman concepts; surveillance and communication. For surveillance, an electro-optical payload is described that could be used for missile surveillance, theater targeting or weather data using the same optics, focal plane and processor. The satellite orbit selected dictates which mission is performed. For communication, both SHF and EHF payloads are defined to provide theater coverage for the tactical user. The advantages and penalties that accrue to the use of a common bus are also explored. In addition, launch options are identified and a comparison made between "launch-on-demand" and "launch-on-schedule" strategies. Potential timelines for rapid launch are shown based on parallel processing and checkout of spacecraft and launcher. This technique is compared with launching satellites on a routine basis and storing them in orbit. Energy requirements for repositioning these stored satellites after they are activated in time of need are defined.

DISCUSSION

The purpose of this paper is to put into context the role of the tactical satellite and present some sample applications in order to provide an introduction to the more detailed design and operations papers to follow. The Tactical Space System, commonly referred to as TACSAT, was envisioned as providing a space capability directly under the control of the military field commander. When needed, the spacecraft can be quickly assembled to provide the required mission capability and quickly launched or repositioned in orbit to provide the required coverage. The system would be compatible with, and operate within the larger in place space architecture, or as a stand alone capability. In either case, however, its operation would be transparent to the user; that is, the user would be able to use the same ground terminals already in use for interaction with the larger backbone satellite systems. The need for a TACSAT can arise from several circumstances. First would be to augment the existing space infrastructure in order to accommodate changes in requirements that could be caused by changes in the military alert posture, e.g., as we go from peace to crisis to war. TACSAT could also be deployed if areas of military instability shift from one geographical location to another. It may, in fact, become necessary to cover several geographical locations simultaneously. During Desert Storm, for example, space assets were repositioned to support operations in the Persian Gulf. Had a crisis or conflict occurred in a different part of the world at that same time, we would have been hard pressed to support operations in both theaters. As I will show later in this paper, space launch systems are not very responsive, nor are satellites stored in orbit generally designed to be maneuvered rapidly. If a failure occurs in one of the backbone space systems, then TACSAT could be used to provide an interim capability until that backbone constellation can be restored. This situation occurred several times in the U.S. military space program, particularly when we have gone from one model satellite to a redesigned or upgraded one. If one looks at an actual supply/demand curve it can be seen that for the sake of economy, most space systems are designed to provide a nominal capability. In general, this capability is less than the peak demand requirements. Also, as mentioned earlier, the capability may be reduced due to system failures. When a crisis occurs, the shortfall between demand and supply could be quite significant.

For TACSAT to fill this gap, it must possess three main characteristics; flexibility, responsiveness and affordability. The first criteria for TACSAT, flexibility means that it must be capable of supporting multiple mission areas. These mission areas are generally defined as surveillance, communications, navigation and environmental monitoring. As I will discuss later in the paper, there are several ways to design a system that is capable of performing more than one of these missions. If we can achieve this flexibility, the number of TACSATs to be built will be maximized, and accordingly, the unit price will be minimized. Another element of flexibility, alluded to earlier, is compatibility with the existing infrastructure. From the logistics and well as an economic standpoint, no TACSAT specific data receiving or processing equipment should be required. This is true not only for the user equipment, but for the facilities required to control and monitor the space systems as well. The final flexibility criteria is launch resiliency. Currently, the military space launch strategy does not include "launch through failure". When a launch failure occurs, a significant downtime is generally experienced in order to troubleshoot and conduct the analysis necessary to determine the cause of the launch failure. Failure is rarely accepted in terms of its statistical probability but rather, because spacecraft and launch systems are expensive, the financial risk attendant to the next launch warrants a thorough failure investigation. The TACSAT concept, on the other hand, is premised on quick response and low cost.

Responsiveness is the second characteristic that a TACSAT system must possess. Responsiveness can be achieved in two ways. The first is to store spacecraft and launch vehicles on the ground and then launch rapidly when the need arises. Depending on the location of the launch site, it may be possible to launch into the inclination of interest and thereby obtain pertinent data on the first orbital pass. Another means of obtaining responsiveness requires the capability to reposition satellites already in orbit. This is a particularly attractive option for satellites in the geostationary belt.

Of all the TACSAT characteristics, the one that is absolutely essential is affordability. If a low unit cost can be achieved the TACSAT concept can be an attractive option especially in this era of limited defense spending. TACSATs can be incrementally acquired allowing the user to purchase the capability needed at present and then add to that capability as the need arises.

One method of achieving affordability is to maximize design commonality across the spectrum of missions to be performed. In the ultimate, one would desire to have a single satellite design capable of performing any mission. This, of course, is not possible. This paper will discuss, however, combining similar missions. A second level of commonality would be to have a common bus and bus subsystems such as power, attitude control, and thermal protection. In this concept, the payloads would be different for each mission. The least degree of commonality would be achieved by having a common bus with subsystems and payloads tailored to the individual mission. Any amount of commonality will result in a larger unit buy, thereby amortizing the RDT&E costs over a larger base and taking advantage of the production learning curve to reduce the unit cost. Second, affordability can be achieved by using the existing infrastructure. Operating with TACSATs should be transparent to the user: it must use the same ground terminals as the backbone space architecture. Also, because TACSATs will tend to be smaller and more proliferated than the backbone system, it will be necessary to make these systems more autonomous thereby reducing the need for satellite control and the costs attendant to that function. For those systems employing the rapid launch option, satellite to launch vehicle integration and test will have to be simplified to allow operation by military personnel without contractor support. Finally, the largest cost driver to space systems are the requirements themselves. As mentioned earlier, the TACSAT concept will allow the user the option to buy only that capability that is needed; the more capability bought, the higher the cost. Other factors that will tend to reduce cost are limited coverage (theater rather than worldwide) reduced lifetime, and a minimum of extras, such as heroic survivability.

A number of potential TACSAT missions were studied. These included surveillance, communications, environmental sensing, nuclear detonation detection, space surveillance and space experiments. The first two missions were chosen to be discussed in further detail in this paper because they illustrate two levels of commonality than can be achieved. The use of a single satellite design to perform two different surveillance missions, theater surveillance and tactical missile tracking, was explored. Although these missions have different requirements, it will be shown that by selecting the proper orbits, both missions can be met with a common payload design. Theater surveillance involves looking at relatively small target areas in order to do target location and then bomb damage assessment. These parameters are also required to allow the user to monitor the battlefield in order to determine deployment and strategies. To do this with reasonable size optics requires that the satellite be flown at relatively low altitude. A 500 km circular orbit was chosen for this application. Tactical missile tracking, on the other hand, requires coverage over a larger area and the ability to detect and track missile launches from the infrared signature given off by the rocket plumes. For this application, satellites in geostationary orbits were postulated.

The theater surveillance system uses an electro optical payload in the visible region for imaging of the selected target areas. At 500km altitude, the payload would be able to acquire targets within an area of 2000 km in track and 1000 km cross track. Within this acquisition basket, the payload would be directly commanded by the user to image target areas up to 9 X 9 km. The data taken by the satellite would be transmitted directly back to the user who would have the capability to do data exploitation in near real time. It is envisioned that the entire process from tasking of the satellite, acquisition of the data and downlinking to the user would take a minimum of 15 minutes. The maximum timeline is governed by the revisit time and is a function of the number of satellites in the constellation and the orbit inclination. The theater missile tracking system, deployed in a geostationary orbit, uses a scanning infrared sensor to detect tactical missiles and after-burning aircraft in a 2000 x 2000 km area. Within that target area, the system is capable of processing up to 1000 potential targets and, after processing, tracking up to 100 real targets simultaneously. For ballistic missiles, both launch and impact points can be predicted. These predictions could then be used to initiate a counterforce strike or cue defensive systems.

A single sensor that could do both the theater surveillance and tactical missile tracking missions was conceived and is shown in Figure 1. The sensor has common optics for both missions and a dual focal plane array to accommodate both the wide field of view and the high resolution requirements. The payload was compact in design and weighed approximately 100 kg. It was now possible to satisfy one of our affordability criteria; a single satellite that could do more than one mission depending upon the orbit into which the satellite was deployed. Once the spacecraft and its payload had been sized, a study was conducted to determine whether the spacecraft could do missions other than those for which it was designed. Figure 2 shows that the missions of multi-spectral imaging and space object surveillance could also be done from a satellite in a 500 km orbit and that the mission of cloud and ocean imaging could be done from geo. The imaging missions would utilize both the IR and visible spectrum at the discretion of the user depending upon the operational and environmental conditions at the time. The surveillance of space objects from space would be done solely in the visible band. To summarize the potential for commonality in the surveillance area, preliminary analysis has shown a minimum of five missions that could be accomplished by a single satellite design. It is expected that a more detailed analysis could surface even more potential missions.

As noted earlier in this paper, a second level of commonality would be to have a common bus capable of accepting interchangeable payloads. To explore this possibility, a second mission area, communications, was selected for study. The study developed conceptual designs of satellites sized to provide direct communications support to tactical commanders. Design concepts at both SHF and EHF were formulated. The tactical users have identified the need for small portable ground terminals that are lightweight, relatively inexpensive, rugged and easily operated. This need for small ground terminals drives the satellite design to the use of high powered transmitters and high gain antennas. The SHF satellite was configured as a bent pipe in which little processing of the signal is done on board the spacecraft. The communications payload simply receives the signal, shifts carrier frequency and retransmits it towards the earth. The EHF system, on the other hand, does much more processing of the signal on the satellite. The signal coming into the satellite is taken off of the carrier and shifted down to baseband where the bits the of digital message are available. The digital bits are routed to their destination, shifted up in frequency, put on the carrier and retransmitted to the ground.

The SHF bent pipe system utilizes two transponders each having a nominal capacity of 80 MHz. A 61 element multi-beam antenna provides anti jam nulling on the uplink and a 19 beam multi-beam antenna shapes the downlink coverage pattern to the theater. In addition, a spot antenna and an earth coverage horn are included in the downlink. Three solid state powered amplifiers are incorporated in the design to provide redundancy. The spare power amplifier can be switched into either of the two active channels. The resultant payload weighs approximately 84 kg and requires 225 watts of power to allow link closure with man portable terminals having an antenna on the order of 0.6 meters in diameter. A capacity of approximately 2000, 2.4 kbps channels would be possible using those man portable terminals.

A 36 channel EHF payload was also sized. This payload was designed to support EHF man portable terminals. The payload consists of 32 low data rate communications channels and 4 channels for noise characterization/acquisition. The sample payload has a 61 beam multiple beam antenna with a nulling processor on the uplink. The fully autonomous operation of the processor represents the only design area that may be pushing the state of the art. The downlink includes a 19 element MBA, a spot antenna and an earth coverage horn. The design features fully redundant travelling wave tube amplifiers. Assuming there is one user per terminal and an average call duration of 4 minutes, the number of terminals that can be supported can be calculated using message switching theory. With a 5% probability of call cancellation or a 20% probability of call delay, approximately 400 to 500 user terminals can be supported by this 36 channel system. The resulting payload weighs approximately 120 kg and requires 245 watts of power.

In the previous section, we sized 3 payloads; a single payload that can accomplish two surveilliance missions, an SHF communications payload, and a EHF communications payload. Payload weights ranged from 84 kg to 118 kg and the power requirements were from 225 watts to 300 watts. If we now try to design a common bus, one that accommodates any of the three payloads, we find that the payload governing the design is that of EHF communications. It is the heaviest payload, has near maximum power requirements, a 10 year life and requires 45 kg of maneuver propellant (the reason for which will be discussed later). The resultant spacecraft weight, including payload, is 635 kg. In comparison with a unique spacecraft designed for each specific mission, the common bus spacecraft represents approximately a 30% weight overdesign for the theater surveillance mission and a 7 to 8% overdesign for the theater missile tracking mission. This basically comes about from the differences in satellite design life which governs the amount of propellant that must be carried for station keeping. In addition, the theater surveillance mission is conducted from lower orbit and does not require the 45kg of maneuver propellant. In comparison to a satellite specifically designed for the SHF communications mission, the common bus design represents a 27% over design. This is mostly due to the lighter SHF payload weight and reduced power requirement. Penalties of this order of magnitude must be accepted to take advantage of a common bus design. Not only does commonality achieve cost reductions as a result of an increased production buy and corresponding learning curve leverage, but it promotes the use of standard test procedures and test equipment. Payloads can be handled as black boxes and thereby, integration and test times can be reduced. It is clear, however, that the more the payload weights and mission parameters diverge, the larger the penalty that must be paid by using a common bus.

The surveillance and communications missions were then used to define the more complete set of bus design parameters shown in Figure 3. As expected, these parameters vary both as function of mission and orbital parameters. In the area of Guidance, Navigation and Control (GN&C) the most stringent requirements (pointing and jitter) are dictated by the electro-optical mission from GEO. This mission also has the largest communications data rate demand. The requirement for autonomy falls under the general heading of Command and Data Handling (C&DH) and can be up to 180 days. To achieve this, it is thought that connectivity with the U.S. Global Positioning System (GPS) would be required to provide ephemeris updates. The propulsion requirements are driven by the need for orbit reconstitution (on-orbit maneuvering). This will be discussed later in this paper. Finally, the bus will need to be protected from the natural space environment as a minimum. It is recognized that a truly common bus design may compromise these requirements but to determine the extend of such compromise will require more detailed study.

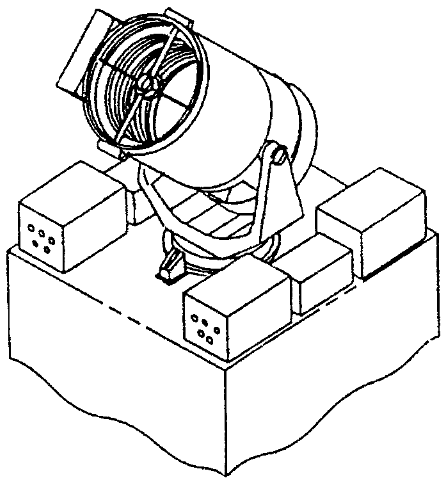
Two launch strategies have been considered for TACSAT application; launch on demand and launch on schedule. To understand the implications of these strategies, the launch vehicles available to the TACSAT must be examined. In the current fleet of United States launch vehicles, the medium launch vehicle (MLV), i.e., Delta 7925, is the one that comes closest to satisfying the TACSAT requirements. With a solid propellant kick motor, it is capable of placing approximately 900kg into a geosynchronous orbit. This represents a 40% over capacity for the 635 kg TACSAT described. An Atlas class MLV, will place approximately 1500 kg into GEO allowing TACSATS to be launched two-at-a time if such a launch strategy is deemed to be advantageous. For low altitude satellite deployment, the Pegasus lift capability is about 400 kg; somewhat shy for the satellite discussed in this paper. The Taurus, which is essentially a Pegasus on top of a Peacekeeper first stage, appears ideally suited to this application. This vehicle is, however, still in the development stage.

Responsiveness is a characteristic generally associated with TACSATS. The capability to rapidly deploy the satellite to the area in which it is needed is essential. When examining the responsiveness of our current launch fleet, however, the nominal time from the mating of the spacecraft to the launch vehicle until the launch can actually take place is 24 days for an Atlas II and 7 days for a Delta II. By streamlining the process, it may be possible to reduce this time down to 7 and 5 days respectively. Add to this the travel time to orbit and the spacecraft checkout time once orbit has been achieved and it is not clear whether the GEO based satellites can be responsive enough using a launch on demand strategy to meet user requirements. On the other hand, for the low altitude satellites launched on a Taurus, it appears that, with judicious satellite design, response times on the order of 24 to 48 hours may be possible. It should be noted that modifying the launch vehicle alone is not sufficient for rapid response. Today's spacecraft can require days or weeks of checkout after they achieve orbit. If surveillance data is to be obtained in the first orbit, for example, design features such as blowdown focal plane coolers and optics contamination avoidance systems must be incorporated.

For the GEO based satellites, an alternate means of providing responsiveness has been studied. In this strategy, the satellites would be launched when available or on some predetermined schedule and stored in orbit. The satellites could be stored in a dormant condition and activated when needed, thereby, minimizing satellite life degradation. The satellites could be stored at a convenient point in the GEO belt and repositioned to the area of interest as the need arises. Figure 4 shows that a 600 kg class satellite could be shifted up to 30° per day with the expenditure of no more than 45 kg of fuel. It is doubtful, however, that if several satellites are stored in this manner, there would be a requirement for this high rate of orbital shift. It seems more reasonable to anticipate maneuvers on the order of 5° per day considering that crises or conflicts do not normally occur instantaneously but rather develop over some period of time. Under these conditions, the 45 kg of propellant could provide 4 to 5 such maneuvers per satellite without affecting satellite life on orbit.

In summary, this initial study has shown that TACSATs can be used to augment the existing backbone space architecture by providing a capability that currently doesn't exist, such as tactical surveillance, or by adding to an existing capability, such as communications, in times of crisis or conflict. In this manner, it could also be used to provide an interim capability should one or more of the backbone satellites fail. In order to make the concept practical, however, the systems must be affordable. One method of achieving such affordability is through maximization of commonality. It has been shown that commonality can be achieved at least on two levels; a single satellite that can perform more than one mission or a common bus with interchangeable payloads. Responsiveness, which is another key element of the TACSAT concept, can also be achieved in several ways. For low altitude satellites, rapid launch on demand is possible while for GEO satellites, storing on orbit and on orbit repositioning appears to make more sense. To make either of these strategies work requires that the system has a low infant mortality, i.e., when you turn it on, it works. Finally, the system must be responsive to user demands. This means user control of the asset and direct transmission of data to the user terminal.

In conclusion, the timing is right for the consideration of a TACSAT capability. With the recent geopolitical upheavals, the focus shifts from the anxiety of global nuclear war to regional, tactical areas of conflict. Such a shift leads to increased demands for information and capabilities that can only be achieved from space. Furthermore, the areas of operation, although limited in size, are likely to be worldwide. The ability to bring assets to bear rapidly will be of paramount importance. Recent experience in Desert Storm has attested to this supposition. The value of space assets for surveillance, communications, weather and navigation was clear. System shortcomings, such as the inability to get some data directly to the user was also evident. Desert Storm also demonstrated the impact of a cooperative, coordinated, multinational effort. This trend is likely to continue in the future forcing requirements to be specified on a universal rather than a national level. These common concerns and needs, along with the severe military spending cuts that are facing individual nations, provide a greater opportunity for international cooperation in the development and use of space systems. The TACSAT concept is particularly attractive in this regard by providing the means of acquiring incremental capability on an as needed basis. If the degree of commonality and interchangeability discussed in this paper can be achieved, the TACSAT can provide a new way of deploying and operating space assets, one that gives the user direct control and the ability to receive critical data in a direct and timely manner.



Optics	• 45 cm aperture
Focal Plane Arrays	• Visible and IR • Wide FOV focal plane - 3° x 0.5° F/7 off-axis • High resolution focal plane - 0.16° x 0.16° - F/15 on-axis
Pointing and Scanning	• Reactionless drives • ± 45° conical field of regard • Wide area coverage in 3° swaths
Envelope	• 70 x 100 x 100 cm
Weight	• 112 Kg
Power	• 370 w (600 w peak)
Downlink Data Rate	• 0.03 - 308 Mb/S (Mission dependent)

Fig 1. Multi–mission sensor

Mission	Theater Surveillance	Multi-Spectral Sensing	Space Object Surveillance	Cloud/ Ocean Imaging	Tactical Missile Tracking
Orbit, km	500	500	500	35750	35750
Spectral Bands, μm	0.45-0.90	Various Commandable 0.4-10.0	0.45-0.90	Various Commandable 0.45-10.0	2.7
Field of view/ scan rate, data rate	1.6° x 1.6° 274 Mbps	3° x 0.1° swath 2.5°/sec 274 Mbps	360° swath 1.8°/sec 20Kbps	18° x 18° 8°/sec 70 Mbps	3° x 3° 8°/sec 20Kbps
Re-visit	12 Hours for one satellite		–	Continuous over Area of Interest	2 sec

F'ı 2. Multi–mission sensor performance

ORBITS	LOW-MEDIUM EARTH ORBITS		GEOSYNCHRONOUS EARTH ORBITS	
MISSIONS / PAYLOAD TYPE	ELECTRO OPTICAL (< 1000 nmi)	OTHER COMM (11000 nmi)	ELECTRO OPTICAL	COMM
BUS PARAMETERS		NAVIGATION		
GN & C				
Stabilization	3-axis	3-axis	3-axis	3-axis, dual spin
S/C Pointing (deg)	.5 to .01	.5 to .1	.2 to .01	.5 to .02
S/C Jitter (deg)	.1 to. .001	.5 to .02	.1 to .0001	.5 to .03
Slewing (deg/sec)	3.0	None	None	None
COMMUNICATIONS				
Frequency Band	S, X, SHF, EHF	S, L, UHF	X, SHF, EHF	S, X, SHF, EHF
Downlink Rate	up to 274 Mbps	4Kbps	up to 274 Mbps	10 Mbps
Downlink BER	< 10-6	< 10-6	<10-6	<10-6
Contact with grd	2 to 12X / day	Intermittent	Continuous	Continuous
C &DH				
Bulk Storage	2 Gigabits	No	No	No
Processing				
Autonomy		180 days	1 - 3 Months	90 days
PROPULSION				
Stationkeeping	No	Yes (+ - .10 deg)	Yes (+ - .10 deg)	Yes (+ - .10 deg)
Alt Adjustment	Yes (+- 25 nmi)	No	No	No
Orbit Reconstitution	Yes	No	Yes	Yes
Momentum Mngmt	Yes	Yes	Yes	Yes
POWER				
EOL Ave Watts	600	< 1000	< 1300	< 1000
Eclipses	Frequent (> 50000 cycles)	Frequent	Infrequent (~3000 cycles)	Infrequent (~ 3000 cycles)
ENVIRONMENTAL				
Rad Dose - Rad (Si)	<500 / yr	5000 / yr	2000 / yr	2000 / yr
Thermal Req's	Stressing	Mod to low	Low	Low
Outgassing	< 50 PPM	< 50 PPM	< 50 PPM	< 50 PPM

Fig. 3. Generalized design budget for TACSAT common BUS

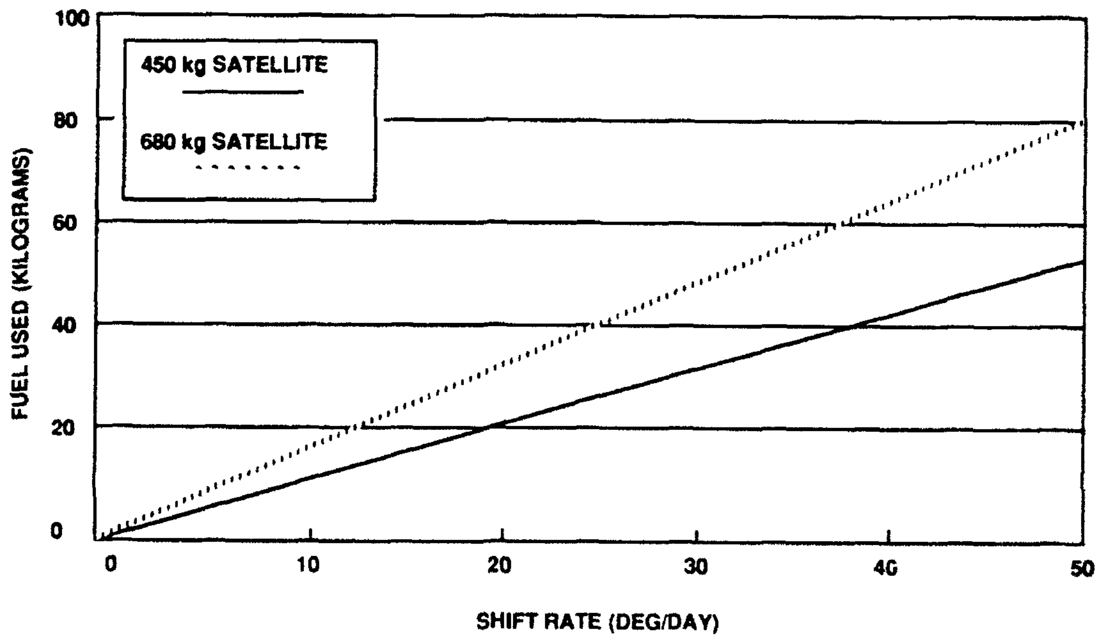


Fig 4. Repositioning at GEO

Discussion

Question: To meet stated requirements for area coverage, how many spacecraft will be needed, given a 9km X 9km IFOV? The issue is whether the total constellation size needed will drive costs beyond affordability.

Reply: A single satellite takes 9 X 9km snapshots anywhere in the acquisition area. The number of areas to be imaged and the dwell time on each target area, as well as the revisit time requirements and the orbital altitude, will dictate the number of satellites required. Tradeoffs have not been conducted to assess affordability as a function of requirements, but, as you observed, the requirements must be kept under control or affordability will be lost.

EVOLUTION OR REVOLUTION – THE CATCH 22 OF TACSATs?

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1 Introduction

The potential exploitation of TACSATs is limited by a vicious circle in which users do not specify requirements which they believe to be infeasible and the space industry does not offer radically new solutions because it perceives no demand. This is the Catch 22 of the title.

This paper aims to show some of the possibilities if we break out of that vicious circle. A private civilian Earth observation mission (SeaStar) is adopted as a baseline. In order to illustrate what might be possible, military payloads for surveillance, verification and C3I are suggested, derived from land or air based systems that either already exist or are known to be under development. It must be appreciated that these are not presented as proposed designs, merely as a kind of "existence proof" for high performance and low cost satellites.

It is possible, on the basis of this "existence proof", to explore some of the operational consequences of the more general use of TACSATs.

2 "Traditional" space thinking

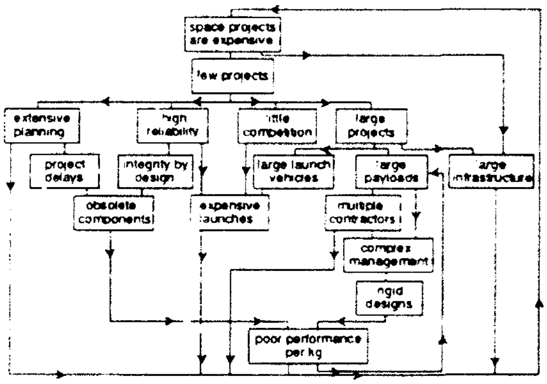
There has been a development in the space projects carried out in the USA and Europe from the early days of simple, dedicated and inexpensive missions, through to the present position where most missions are complex, multi-purpose and expensive.

This is a result of a positive feedback mechanism[1] which systematically forces space missions to become more expensive, take longer to execute and fail to satisfy the needs of their users. The positive feedback mechanism starts with the belief that space projects are expensive. The consequence of this belief is that there will not be many projects. Few projects mean that:

- they must be planned carefully to get the best out of them;
- they must be reliable;
- they need to be large to achieve a lot from each project;
- there will be little competition (not only commercial competition between suppliers but there is also little room for competition of ideas).

Each of these brings consequences for the conduct of the projects. Planning causes delays. High reliability when not building many systems means that integrity must achieved by design. This precludes the use of the latest, unproven technology and, taken with the planning delays, means that spacecraft are built with obsolete components. Large projects need large launch vehicles which, when combined with the lack of competition and the need for high reliability, means that launch is expensive.

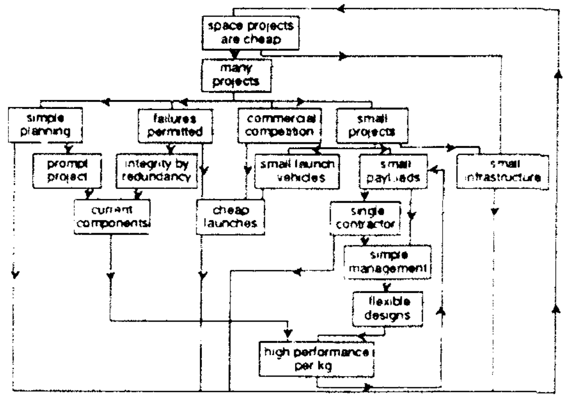
Large projects also require large payloads and justify a large (and expensive) ground infrastructure. Large payloads bring a twist of their own. The space systems are too big to be built by a single contractor and it is necessary to invoke a complex (and expensive) management structure. One consequence of this is that it is difficult to perform optimisation trade-offs between components of the system because of the rigid contractual boundaries. This results in non-optimum designs built, as argued above, with obsolete components, which leads to very poor performance per kilogramme. Here is the first vicious spiral because poor performance per kilogramme requires even bigger payloads.



When all of these arguments are brought together, the resulting vicious spiral reinforces the opening premise:- "space projects are expensive". Any perturbation, like a launch vehicle failure, causes more passes around the spiral which reinforces the premise and the costs climb further.

It is important to recognise that there is no malicious act in this spiral. It is a consequence of reasonable decisions being taken at each stage of the design and planning of these projects. That is why revolutionary changes in the procedures, practices and thinking are needed to reverse the cost drivers.

What could happen if all of these changes in approach were to occur? If it is assumed that space projects are cheap, a different positive feedback picture emerges, a virtuous spiral.



This now points towards the revolutionary approach. Miltary space missions could exploit sub-systems already developed for air or land based use and achieve integrity through redundancy (many missions) and through the benefits of a production run rather than one-off build. The reduced cost of the missions would allow them to be treated as tactical, rather than strategic, assets. The rest of this paper will look at some possible military missions for surveillance, verification and C3I, based on a standard civilian Earth observation bus.

3 Baseline for military missions

Background

The SeaStar mission, being executed by Orbital Sciences and Hughes Aerospace provides a reference on which to base possible military missions[2]. It is a civilian earth observation mission, designed to monitor the ocean colour, a measurement that is considered to be of great value to environmental research and possibly to be of commercial value.

SeaStar is based on the PegaStar bus which offers:

- 3 axis stabilisation to $\pm 1^{o}$;
- at least 170 W mean electrical power;
- up to 5 year life;
- up to 70 kg payload mass;
- payload <1m diameter and <1.5 m long;
- encrypted data downlinks (L-band and S-band at up to 2 Mbit s^{-1});
- >150 Mbytes on-board data storage;
- shock and vibration environment similar to that of military aircraft

Cost

The financial basis of SeaStar is:

- commercial private venture development and operation, against commitment by NASA to purchase data ;
- agreed price $43M for 5 years of data, including all construction, launch and operations.

It will be assumed that any of the payloads considered in this paper will cost no more than the SeaStar payload. On this basis, it will be assumed that the total cost of a 5 year mission, including launch and operations, will be $10M per year. Informal discussions with Orbital Sciences have indicated that a short mission (<1 year) would cost no more than $12M.

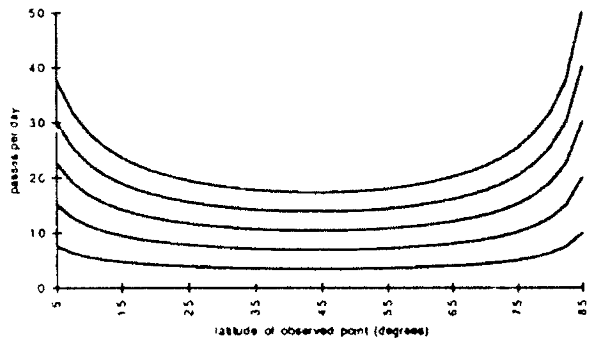
Orbital altitude

All of the payloads considered in this paper are more effective at lower altitude and 400 km will be assumed. Drag from the residual atmosphere will be between 0.02 mN m^{-2} and 0.' mN m^{-2}, depending on the state of the solar cycle[3]. Orbit maintenance will require, at worst case, 2 kg of fuel per year, assuming a cross-sectional area of $2m^2$ and a thruster of I_{sp} = 300 s. It will be assumed that such a spacecraft at 400 km altitude will require no more fuel at launch than would be needed for SeaStar at 705 km.

Observation opportunities

It will be assumed that the purpose of the spacecraft is to observe a specified point as frequently as possible and, for simplicity, only circular orbits will be considered.

The average number of passes per day by 1 to 5 satellites in the most appropriate orbit at an altitude of 400 km is shown below, assuming that the sensor has an off-nadir capability of 300 km[4]. The distribution of those viewing opportunities depends on the exact choice of launch time and inclination of the orbit.



4 Possible military missions

Three types of mission will be considered: - surveillance, verification and C3I. In each case, the approach will be to examine the performance that might be expected if the payload were to be a conventional terrestrial or airborne system.

The mechanical stresses of launch will be no worse than encountered in a military aircraft. Radiation effects will not be significant at an altitude of 400 km. The only special constraint of space is the need for high reliability - 40,000 hours would be required for a 5 year mission.

Surveillance

A possible surveillance payload is the airborne radar under development for the Advanced Tactical Fighter radar[5]. Annex A shows that it is falls within the constraints of the mission baseline.

Three surveillance modes can be considered:

- for air surveillance:
  - look-down Moving Target Indication (MTI);
  - matched illumination;
- for ground surveillance:
  - spotlight Synthetic Aperture Radar (SAR).

The application of these in naval or land actions can be seen as analogous to stand-off radar, Remotely Piloted Vehicles (RPV) or the ideas for organic air protection of fleets using airships.

A different approach to surveillance would be to use an infra-red imager, operating in the 3-5μm or 10-12μm bands. Annex B describes an optical sensing system for the visible bands and informal discussions with Questar have indicated that an IR version of the 12" telescope is being considered. A 256 element linear detector array would allow imaging a strip ~2.5 km wide with ~ 10 m spatial resolution in the 3-5μm band. This would allow detection of, for example, thermal signatures on runways where aircraft have taken off or the detection of hot spots on armoured vehicles caused by the heat of their engines.

Verification

Space-based verification systems may be used to cue ground or overflight inspections. A system with moderate spatial resolution but with the flexibility to image frequently and without warning might be an effective deterrent to breaches of a treaty.

The optical system described in Annex B would allow 2 km square images of any point to be obtained with a spatial resolution of 1 m. This is adequate for the detection of all and recognition of most types of target. The satellite may be tasked with a list of points of interest and can then autonomously collect the images and either broadcast them to a local receiver or store them on board and down-load when next passing over headquarters. There are several opportunities per day to image the point of interest with one satellite.

C3I

There are many aspects to C3I which might be addressed by means of satellites. Annex C considers the feasibility and performance of a radar ESM system used to detect and identify pulse and CW emitters. The baseline system, Kestrel, is designed for airborne use and can offer significant tactical capability, particularly if the satellite version of the multi-port antenna has higher gain than that employed in the airborne version.

One possible use of this system would be tactical ESM. The system on-board the satellite would include the capability to deinterleave and track radar emitters and would broadcast its track table (possibly including radar identification) to tactical field units. Although ESM only gives an estimate of the direction of the threat from the receiver but not its range, the rapid motion of the satellite might make it possible to locate the transmitter by triangulation as the satellite passes. The use of two or more satellites has not been considered here but clearly this could provide more immediate and reliable location.

An even more powerful way of using two satellites cooperatively would be to compare the time of arrival of individual pulses at each satellite rather than to compare the characteristics of pulse trains. This would not however be within the capability of a conventional airborne ESM system like Kestrel.

5 Operational implications

The possible military missions suggested indicate what might be possible. There are two important operational implications:

- it is assumed that the satellites would be launched on demand to provide cover of specific points of interest;
- the satellites can communicate directly with field units, allowing the data that is collected to be relayed in real time to users and allowing the users to task and control the satellites.

Launch on demand of a constellation of satellites may seem to be an expensive option. However, it should be seen in the context of the costs of conventional surveillance assets such as stand off surveillance aircraft. It is clearly not appropriate to compare directly the costs of TACSATs and aircraft since they offer different capabilities but surveillance aircraft illustrate the sums of money that can be allocated to tactical surveillance, verification or C3I.

The cost of operating a surveillance aircraft is of the order of \$5000 per hour. Thus, the cost of 24 hour cover for one year would be of the order of \$40M. To this must be added the effective cost of the finite probability that the aircraft may be lost due to enemy action.

A constellation of 5 satellites would provide cover many times per day and not be vulnerable to air to air or ground to air missiles at a cost of no more than \$60M for one year. The financial case for satellites is stronger if the operation is required to continue for longer than one year. For example, a drugs interdiction support operation could use 3 satellites to view Central America and the northern coast of South America more than 15 times per day at a cost of \$30M per year.

Direct communication with field units represents a revolutionary change in the operational management of space assets. It recognises that TACSATs are tactical assets, to be deployed and exploited under the control of tactical commanders analogous to any other tactical asset.

It is inconceivable that space assets could be considered tactical unless they were much cheaper to purchase and operate than than current strategic assets. However, it is exactly this approach, which would lead to production runs of satellites comparable to those of, for example, fighter aircraft, that would cause the space assets to be cheaper.

6 Conclusions

The argument has run full circle. This paper has presented an approach to the provision of TACSATs based on three premises:

- the use of an inexpensive standard bus;
- payloads using military sub systems;
- direct control and tasking of the satellite from tactical commanders in the field.

If these three rules are adopted, it is argued that it will be possible to break out of the Catch 22 and achieve inexpensive operational TACSATs which convey significant military advantage.

7 Acknowledgements and references

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Annex A Radar system

A.1 Background

It is possible to establish the feasibility in principle of operating a radar on a TACSAT, by examining in outline the power budgets and hence signal/noise ratio that might be achieved using an advanced airborne radar. The model that will be used here is the Advanced Tactical Fighter (ATF) radar currently under development. A description of the this development programme[5] included the following quotations:

- The USAF has pursued X-band AESA [Active Electronically Scanned Array] technology since the early 1980s for three main reasons: power/weight ratio, agility and reliability. With more than a thousand transmit/receive (T/R) modules, each capable of generating around 10W of power, the F-22 has a peak power in the megawatt range, should that be required.
- Ultimately, a maintenance-free life of 20,000 h - the lifetime of the aircraft - is possible.
- Synthetic aperture radar (SAR) ... is not in the baseline but the hardware can do it ...

A.2 Radar equation

The following assumptions will be made:

- number of T/R modules: 4000
- module power: 10 W
- square array, half-wave element spacing
- wavelength: 30 mm

This indicates an antenna gain of ~39 dB and an effective antenna area of ~ 1 m^2. If the spacecraft is at 400 km altitude and a target of cross section 1m^2 is located 300 km from the satellite track, then the received power will be -165 dBW. If a receiver noise figure of 5 dB and detection threshold of 10 dB are assumed, then the integrating time will need to be ~ 3ms.

The beam width will be ~ 15 km at 500 km range. An area of 500 km square will comprise of the order of 1000 resolution cells. A transmitter power of 40 kW and integrating time of 3 ms per cell will require ~ 0.1 MJ of transmitted energy. The total energy available to the payload is of the order of 50 MJ per orbit.

A.3 Operational modes

Air surveillance

Any look-down radar suffers from ground clutter and the system described here will be particularly badly affected because of the relatively wide beam width. Two techniques that might be employed to improve the detectability of airborne targets are:

- Doppler MTI;
- matched illumination.

Doppler MTI may be employed provided that the radar is directed across track to detect targets with a significant velocity component towards the satellite. The performance of MTI with the wide beam of this radar needs further analysis.

Matched illumination uses modulation schemes matched to the characteristic dimensions and resonances of the target. This has been used to identify specific aircraft types[6]. No trials have been reported of the use of this technique to reject ground clutter but the selectivity shown in the identification trials suggests that it might be effective.

Ground surveillance

Ground surveillance to detect and locate vehicles could be carried out using spotlight SAR. Typical performance would be to form a SAR image of a single 15km diameter region (corresponding to the beam width) on each satellite pass. A synthetic aperture of 15 km (corresponding to 2 s of illumination) would allow a SAR image of 1 m resolution to be constructed.

Current real-time SAR processors capable of this level of processing have a mass and power consumption suitable for airborne use and might feasibly be carried on-board the satellite. Alternatively, the raw data could be broadcast for ground processing.

Annex B Optical system

B.1 Background

This design explores what might be achieved using existing space or military quality subsystems to build a high resolution imaging system. The spacecraft has a space-qualified version of a commercial telescope mounted such that it can be rotated about an axis aligned within ~ 1^o of the direction of travel. A linescan imager is mounted at the focus of the telescope, aligned so that it sweeps a swath along the direction of travel (pushbroom).

The design presented here can provide 2 km square images with a spatial resolution of 1 m. The image region is selected by rotating the telescope for the across-track dimension and selecting the time of recording for the along-track direction. The position of the satellite must be determined to ~ 50 m in all three dimensions and the orientation of the telescope to ~ 1 mrad in three directions to permit the centre of the image to be selected to an accuracy of ~ 0.5 km.

B.2 Telescope and sensor

The primary optical system is a space-qualified version of the standard Questar 12" telescope. The key parameters of its specification are:

- resolution:	0.38 arc sec
- focal length:	4572mm
- aperture:	305mm
- dimensions:	~1m long, ~350mm diameter
- mass:	~55kg
- vibration tolerance:	10g
- material:	invar
- price:	~$200k

It is understood that this telescope has been used on US military spacecraft.

The baseline design is to use the Reticon RA2048J charge coupled device (CCD) detector operating in time delay integration (TDI) mode. It has to be configured so that its long axis of 2048 sensors is perpendicular to the track of the spacecraft and the clock frequency in the transverse (short) axis of 64 detectors is synchronous with the velocity of the spacecraft.

The signal/noise ratio of the detector, D, is given by:

$$D = r\, a\, A\, T\, s/N$$

where:

r is the radiance of the Earth, assumed to be of the order of $100 Wm^{-2}sr^{-1}$. (This figure is sufficiently conservative to include large sun zenith angles encountered at high latitudes.)

a is the area of ground mapped onto one pixel of the detector (assumed to be $1 m^2$)

A is the solid angle subtended by the telescope aperture (0.45×10^{-12}st at an altitude of 400km)

T is the integrating time, defined as the time taken for the spacecraft to fly 64m (8.5ms)

s is the sensitivity of the detector (480 nV J^{-1})

N is the detector noise level (200 µV RMS)

The signal/noise ratio is thus ~ 1000.

The detector pixels are approximately 25µm square. This requires the focal length of the optical telescope to be 10m. A secondary lens will be needed but this does not have to be of particularly high optical quality. The type of lens used as a x2 teleconvertor for 35mm SLR cameras would probably be suitable.

B.3 Navigation and positioning;

If it is assumed that the nominal 2km square image must be centred on the target position with an accuracy of ± 500m, it is necessary to know the orientation of the telescope to an accuracy of the order of 0.5 mrad.

Navigation information will be derived from a GPS receiver and orientation is obtained from a star sensor.

A possible implementation of the GPS receiver is to carry out the signal acquisition and processing in software within the on-board processing system. A baseline design using a single transputer exists and would meet the requirements with a power consumption of ~0.5W. It is likely that a more appropriate processor could be used to reduce this.

The star sensor is mounted on the telescope to ensure that there is a constant angle between the two. If it is assumed that the properties of the star sensor are:

- field of view:	25^o
- number of stars for reliable fix:	6
(hence need to use stars of 5th magnitude)	
- spacecraft roll rate:	10 mrad s^{-1}
(= ~ 1 revolution every 10 minutes)	
- integration time:	50 ms
- flux from 5th magnitude star:	2.5×10^{-14} W cm^{-2}
- detector sensitivity and noise:	as Reticon above
- sensor aperture:	40 mm diameter
- detector array:	500 x 500 elements

then it will be capable of meeting the target of 0.5 mrad accuracy if it is possible to interpolate to one half of a pixel with a signal/noise ratio of around 70. This is considered to be well within the performance of current interpolation algorithms.

Annex C ESM system

C.1 Background

An airborne ESM system might provide a suitable basis for a tactical communications or radar monitoring satellite. The baseline that will be assumed here is the Kestrel Mk II made by Racal[8]. This is a radar ESM, capable of detecting, deinterleaving and tracking radar emissions between C and J band.

C.2 Detection capability

Kestrel employs a six port amplitude comparison receiver to measure signal bearing. The standard antennas give 360° coverage in azimuth, 25° coverage in elevation, a typical pulse sensitivity of -60 dBmi and bearing accuracy of 4.5° RMS. This would detect the main lobe of a 10 MW ERP C-band radar at a range of 500 km. It would also be possible to locate the radar to an accuracy of approximately 50km by tracking the bearing of the radar as the satellite passes.

Antennas with greater gain could be used at the cost of sacrificing the 360° capability. This would be more easily achieved at higher frequencies and it might be appropriate to replace the Band 3 antenna (I to J band) with a multiple feed horn and reflector. A reflector of the order of 500 mm in diameter might be expected to give 20 dB improvement in sensitivity and a factor of 10 improvement in bearing accuracy but would not add greatly to the mass or cost of the system. This would allow the system to detect sidelobes of pulse radars and to locate them to an accuracy of approximately 5 km.

Deinterleaving and tracking of repetitive pulse trains and identification of emitters is strongly dependent on frequency measurement. The worst case doppler shift caused by the satellite motion will be ~ 1 MHz and hence within the frequency resolution of the receiver.

C3 Mechanical and electrical constraints

The three main building blocks of Kestrel, other than the antennas and receivers, are a parameter measurement unit (PMU), a data processor unit (DPU) and a man-machine interface (MMI). The PMU would need to be on-board the satellite, the MMI is clearly on the ground and the functions of the DPU could be distributed between the satellite and the ground. There would be merit in maximising the fraction that is on-board the satellite because it is likely that there would be many users for one satellite, each of whom would require the ground-based elements of the DPU.

Kestrel is specified to meet the military temperature range and its vibration and acceleration tolerances are adequate for a PegaStar vehicle at launch. The total mass of Kestrel is approximately 50 kg and the power consumption, excluding cooling but assuming that all of the DPU functions are included, is approximately 500 W.

If it is assumed that the system is operational for ~ 10 minutes per orbit, the average power consumption is ~ 55 W, within the PegaStar capability. In continuous operation, Kestrel requires forced air cooling for continuous operation. In intermittent operation with no forced cooling, it is estimated that the average temperature of the equipment will rise by ~ 1°C per minute. If the maximum continuous operation is for 10 minutes, the temperature variations will not exceed 10°C.

If this variation in temperature is not acceptable, it would be necessary to improve the thermal coupling between the heat-generating elements of Kestrel and the rest of the satellite. Heat pipes might be considered or it might be reasonable to contain the Kestrel units in a sealed and pressurised container and to use forced air to redistribute the heat.

TACSAT Ground Control and Data Collection

by

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Abstract

This paper will address the concept of a satellite based system serving the needs of tactical users for direct access to communications and various forms of surveillance. Such a system must take account of the most effective methods for deployment and maintenance during its intended period of operation.

In terms of the mission need the Tactical Satellite System (TACSAT) is required to provide the services for tactical users over a relatively small region of maybe some 1000 - 2000 km diameter. Also the concept is likely to involve relatively short periods of operation of about 3 to 6 months for a typical operation scenario. The paper therefore addresses a system concept in which the emphasis is placed on reducing the scale of the logistics involved in the deployment of TACSAT elements and simplifying the ground operations and facilities necessary for the users to gain access to the services provided.

The paper will address a number of issues which arise such as:

- The need for 'launch on demand'
- The possibility of launches being directly under the control of the area commander
- Operating concepts for TACSATs, comparing the approach of launch-via-residual-mass as part of a larger mission, not dedicated to the TACSAT mission with the approach of dedicated, launch-on-demand.
- Identification of information flow requirements for TACSAT integrity and status evaluation and for surveillance data recovery
- Determination of technical and cost drivers influencing the form and cost of ground installation and logistic issues
- Integration of TACSAT facilities and services with other communications and surveillance systems available to the Tactical Commander
- Investigation of techniques to minimise the ground control and data collection overhead

Introduction

All TACSAT Systems are designed to provide a Tactical Commander with short term operational data in addition to that available via strategic resources. This paper discusses the interfaces between the space and ground segments of such systems. Because there is such variety amongst the various TACSAT systems concepts, it is only possible in a brief paper such as this, to discuss the ground/space interface in general terms. Nevertheless offering a generalised structure around which specific concepts can be detailed is thought to be useful, especially where the concepts are unlike those of other types of space system that designers may be more familiar with. The paper therefore focuses on TACSAT system concepts that are not like classical geosynchronous or sunsynchronous missions.

Mission Definitions

For tactical operations any system employed must be able to support the mobility requirement and operate from unprepared and unsurveyed sites. Communications in difficult terrain, such as mountainous regions, coupled with the need to co-ordinate ground, air and sea forces, presents a complex array of tactical communication needs and information exchange requirements in a variety of forms, particularly for an extensive tactical theatre of operation.

Tactical Imaging missions will be targetted on small, mobile platforms (eg aircraft and tanks) as well as other key targets (eg bridges, harbours). Multiple sensors will be most helpful in penetrating camoflage of these targets. For Tactical Communications Systems, voice and data communications, point to point, point to multipoint and netted arrangements may be all required. The End Users may wish to call upon any or all of these services at any time. In the tactical situation the unpredictability of the operational environment dictates that communications must be instantly available, reliable and trustworthy (in terms of a low probability of detection), have a high probability of timely connection, and have an appropriate level of information security.

It is evident that the introduction of complex procedures for terminal operation, access constraints and rules requiring a high degree of user skill to understand and implement will detract from the usefulness of the system as perceived by the tactical user community.

TACSAT systems can be classified as providing either image data derived from spaceborne sensors or relay communications channels between sites on the ground, or both. Classically these requirements are met by standardised systems architecture as follows:

Image Data	- High resolution, Sun-synchronous, polar, circular
	- Geostationary
Communications	- Geostationary

As has already been stated, it is not particularly useful here to analyse the ground/space interface within such architectures, beyond stating that TACSAT applications in similar architectures are possible, principally because of excess capacity becoming available for tactical purposes within strategic systems. For example, "spare" capacity sometimes becomes available when a spacecraft suffers a partial failure and can no longer meet the full strategic requirement. When it is replaced by a new spacecraft of the standard design, the degraded spacecraft can then be operated as a TACSAT. In another case, a standardised design of geosynchronous satellite has been deployed to three (or more) stations in the geosynchronous arc to provide a global system but strategic communications requirements on one (or more) of the stations do not call for such a large spacecraft. "Spare" transponder(s) on such under-utilised strategic comsats can then be operated as a virtual TACSAT. Finally, it is sometimes the case that "spare" launcher payload capacity into polar orbit becomes available because spacecraft design constraints were too conservative for the actual launcher performance achieved in a parallel development, and small TACSATs can be launched into similar polar orbits with this spare capacity. In all these cases the ground/space interface is similar to, if not identical to, the parent system. In some further cases, very similar mission concepts are chosen for purely tactical reasons. However, in general, different mission concepts tend to be favoured because of the tactical military requirements for TACSAT systems, for example:

Flexibility under unsophisticated, local control

Low cost

Minimum revisit times

Localized area of interest

Surprise

Secrecy

but, most of all,

Ease of use.

Other papers in this series illustrate instances of this tendency. Here, a typical "novel TACSAT" mission may be summarised as:

Intermediate orbit inclination, optimised for target area coverage.

Low altitude, for maximum resolution and/or link margins.

Elliptical orbit, minimising geodynamic drag.

Highly manoeuvrable spacecraft to maintain orbit alignment with respect to the target area.

Pre-programmed payload operation.

Ground Control

During the full novel TACSAT mission there will be the following, distinctly different, phases of operation.

Launch Preparation Phase

It is possible that there will be a choice of Launch Vehicle configuration. For each, there will be a complex trade-off between orbit parameters (especially orbit inclination), payload mass (or functionality) and target area revisit periodicity. There will also be a complex trade-off for the phasing of any particular orbit between spacecraft constraints (eg sun angle at injection), operational coverage requirements and spacecraft fuel consumption during orbital manoeuvres. There may be a military requirement to deceive a sophisticated enemy, able to identify and track novel TACSAT launches, as to the nature of the intended tactical support, especially phasing over the target area. Finally, there may be a requirement to defend the TACSAT system against jamming.

The trade-offs must be resolved and the launcher/spacecraft configuration finalised (including programming of on-board computers) early in the Tactical Deployment when the Tactical Staff will have many other pressures to finalise deployment plans and logistic support. However, since it cannot be assumed that these staff will be familiar with spacecraft operational constraints and speedy decision making may be of the essence, previous in-depth training in system operation will be a pre-requisite of successful TACSAT operations.

Launch and Early Operations (LEOPS) Phase

Launch services would be provided by a specialist supplier requiring minimum interaction with the Tactical Users. The boosted ascent and spacecraft separation will be pre-programmed. However, determination of the initial orbit by ranging during first apogee will be necessary from appropriately located and equipped ground stations. Note that the spacecraft ranging transponders may well need to be protected by encrypters from ranging, and/or from jamming, by enemy ground stations; hence digital ranging, military staff and cypher distribution channels will be preferred For accurate early orbit determination, these ranging stations would be on a long (> 1000 km) baseline and must pass ranging data sets by communications links to a Central Control facility equipped to compute the initial orbit. It will be convenient to arrange for this Centre to also compute necessary manoeuvres, predict the subsequent orbit evolution and, most probably, execute the manoeuvre sequence in a manner consistent with any requirements for payload deployment/activation/calibration. To effect the manoeuvre/activation sequence, Tracking, Telemetry and Command (TT and C) access would have to be provided via several, well-distributed Ground Stations if this phase is to be completed with minimum delay. In short, this entire phase could be under the control of the launch site, provided with support from a network of appropriately equipped Ground Stations.

However, in view of the military sensitivity of TACSAT missions, it is more likely that all Ranging, Telemetry, Manoeuvre Computation and Telecommand functions would be exercised by a suitably equipped and military-staffed Facility. For reasons of compatibility with Launch Site and back-up orbital support, TT and C functions could be conducted at S-Band using the Space Ground Link System (SGLS) standard, but with encryption enabled shortly after successful injection into initial orbit. Ranging, Telemetry and Telecommand functions could be exercised by a 19 m S-Band Telemetry & Command Station (TCS), supplemented by relatively small (3 m) S-Band terminals co-located with selected permanent Ground Stations. Alternatively, the entire TT and C support could be exercised within the X-Band channel allocations for Milsatcom and Earth Resource Downlink channels. In either case, TT and C baseband connectivity during the periods of TACSAT visibility must be provided, for example via orderwire channels within permanent communications accesses.

TACSAT Operations Phase

The TACSAT system is seen in this paper as supplementing strategic C^3I resources, particularly where communications requirements peak or imaging data are needed at short notice. The opportunities to use TACSATS to best effect can therefore only be recognised locally and should put into effect by a local Control Centre.

The Control Centre must be able to assess the effects of orbital manoeuvres, execute manoeuvres promptly and quickly confirm the achieved new orbit. As has already been said, speedy and accurate orbit determination can only be achieved with long, ie out-of-Theatre, baselines. Certainly for imaging missions, perigee will be over the tactical theatre; for communications missions this may also be true.

The out-of-Theatre character of the control function for some missions is reinforced by the need to manoeuvre near apogee, where manoeuvre fuel usage is minimised and visibility to ground stations is maximised. The Control Centre may also have to call on spacecraft equipment designers and test configurations to resolve spacecraft performance anomalies. All of this tends to suggest that the responsibility for TACSAT Control could be assigned to a Military Spacecraft Control Centre supported by a global tracking network but with an excellent co-ordination interface to in-Theatre tactical operations staff. This latter requirement could be met by assignment of several specialists (with recent spacecraft operations experience) to provide a 24-hour service at the Tactical HQ. The Control Centre used during the earlier Phases could be used to execute spacecraft control. However, a knowledge-based system located in-Theatre is likely to provide faster response and more consistent operations.

This could be automated to minimise the number of operations staff and provide consistency during 24-hour operations. It would consist of the following elements:

Automated Mission Planning	Builds timeline options
Automated Operations Planning	Library of Flight Control Plans, including Anomaly Contingency Plans.
Computer Assisted Operations	Executes actual control via Tracking Stations.
Adaptive Training	Creates and evaluates training sessions.

Network Control is also required in the case of a TACSAT communications mission. It can be assumed that the communications transponder will be accessed only by in-Theatre force elements, using locally-assigned cyphers. In order to provide communications services compatible with the needs of tactical End Users, the way in which the network is managed and user access is granted and controlled must involve a minimum of "overhead" workload imposed on the End User. This can range from avoiding the need to point the antenna on his terminal precisely, to the simplicity of operation of the equipment, to a means of establishing user confidence in the integrity of the system, and to its ability to provide the services required. Optimum use of the TACSAT capacity will be most easily achieved by a mixture of frequency assignment and timeline planning issued by in-Theatre signals staff. Some types of TACSAT store-and-forward communications mission will require quite complex space/ground protocols which must be transparent to the End User. Some classes of TACSAT will provide imager or transponder configuration options that are selectable by ground command. The in-Theatre specialist TACSAT operations staff will be best equipped to make the appropriate option selections, which could be implemented either by a local telecommand uplink or by a centralised Control Centre. In the case of in-Theatre jamming attacks, the former is likely to provide a far more successful ECCM response than the latter, provided it can counter any enemy jamming of the telecommand uplink.

The final stage of the TACSAT support will be disposal of the spacecraft, either into a higher parking orbit or a burn-up during re-entry. Once the in-Theatre authorities have confirmed that TACSAT support is no longer required, they need have no further involvement in the manoeuvre sequence. At this stage execution and monitoring of the TACSAT disposal could be turned over to a central Control Centre.

Data Collection

Modern spaceborne imagers produce very wide bandwidth data rates which quickly saturate available spaceborne storage devices. Hence direct-to-Theatre downlinking is very attractive for small TACSAT missions where data timeliness is of the essence. These downlinks must, for obvious reasons, be encrypted. Because of the very high burst rates, spread spectrum downlink jamming protection is not practicable; the only feasible protection methods are Downlink User Terminal (DUT) location and minimisation of DUT antenna sidelobes. Some on-board storage buffers would allow greater flexibility in the deployment, and reduce the numbers, of in-Theatre DUTs. These DUTs must be connected by telecommunications links of various kinds to End Users. On-board preprocessing can be used to reduce the downlink bandwidth and reduce the complexity of the in-Theatre image processing functions. However, some on-ground processing, including fusion with other data sets, will be an essential element of the image assessment chain.

In the reverse direction, the flow of information from the End Users for target selection must converge on at least one Telecommand Uplink facility. Instrument settling times dictate that, for efficient use during each perigee, these telecommands must be uplinked out-of-Theatre during apogee.

Simulation

From the above discussion it is clear that deployment of a TACSAT system is a complex undertaking requiring well-trained operators and well-briefed operations staff. Although the classic military process of reducing complexity down to the minimum number of Standard Operating Procedures (SOPs) will no doubt be applied, this must be complemented by realistic training for all specialist staff and the provision of adequate operations and planning tools. Both must depend on comprehensive mission simulations for the Launch Preparation and TACSAT Operations Phases.

Conclusion

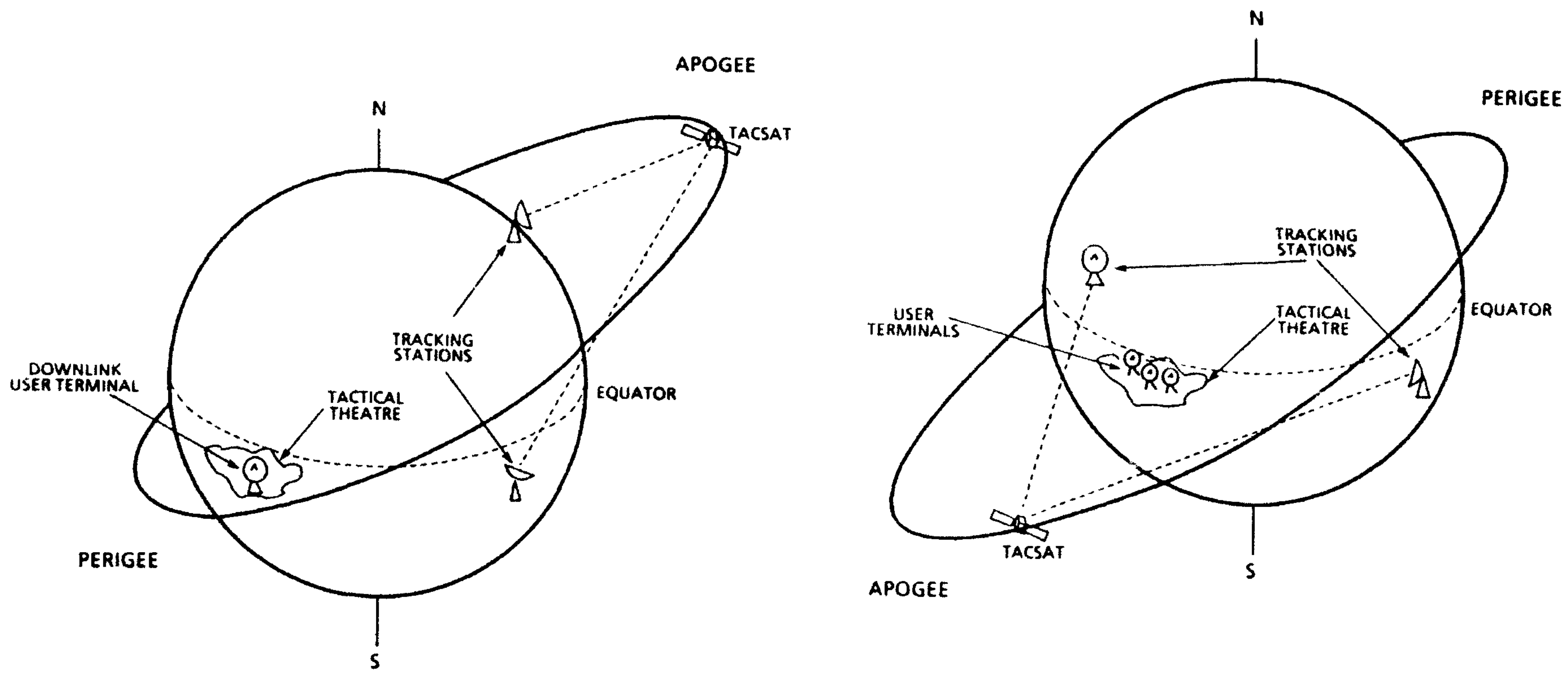
Clasically, Tactical Commanders have succeeded where they have made best use of the time available before tactical deployment and thoroughly prepared to undertake the type of operation to which they were actually assigned. The great advantage of TACSATS is that they are very flexible so it is not necessary to detail in advance the type of Tactical support required beyond a broad characterisation. However considerable expertise is required to operate any Satellite system and any TACSAT Mission is likely to fail unless an adequate control system is created and worked up to a high state of readiness before the Tactical situation develops. An automated knowledge-based system, including appropriate simulation, will be essential to the success of TACSAT support, which in turn is likely to be decisive on future tactical battlefields.

STATEMENT OF RESPONSIBILTY

Any views expressed are those of the author(s) and do not necessarily represent those of HM Government.

TACSATS FOR SURVEILLANCE, VERIFICATION AND C3I

TYPICAL " NOVEL TACSAT " ORBIT FOR



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SURVEILLANCE

COMMUNICATIONS

TACSAT-SPACECRAFT BUS CONCEPT AND DESIGN: APPLICATION OF A MULTIMISSION BUS FOR TACSAT IN LEO

by

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FROM A FEW LARGE SOPHISTICATED SATELLITES ...

Before describing the way, a low earth orbit multimission Bus for tactical application is designed, it is convenient to analyse in some details what is a tactical satellite as far as such satellites are very often associated with the notion of cheap small satellites.

For 30 years the size, the mass, the complexity, and the cost of the satellites have been increasing. Space programmes, especially in the defence area, tend to use heavy and costly satellites in a too small number of specimens compared with the needs of users, they have then to continuously monitor priority conflicts such as : budgetary conflicts to decide which programme has priority, operational use conflicts when programming the missions of so few spacecrafts.

In the today context of budgetary constraints, raising again is the well known syndrome "one aircraft, one tank, one satellite as unique future equipment for defence".

The spacecrafts number limitation is the consequence of an inflationist spiral which can be described in three stages :

a - unit cost increases

b - then for a given budget, number of operational units decreases

c - a limited number of spacecrafts leads to pusch for mission optimisation with :

- an increase of in orbit life time and reliability requirements
- the need to group several missions on the same spacecraft
- technical sophistication
- conservative approach including meticulous development and complex verifications
- no risk approach, as the unit cost of each satellite is such a high stake

All these points lead to unit cost increase and bring back to stage a in the inflationist spiral.

...TO CONSTELLATIONS OF NUMEROUS CHEAP LIGHTSATS

To break this spiral, some experts suggest a "Smart small cheap lightsats" concept. Their main idea is to drastically decrease the unit cost of the spacecraft in orbit, this cost including the satellite recurring cost itself, the launch cost and the ground segment and operations costs.

This concept is consistent with a system architecture using constellations of numerous satellites of limited in orbit life time and low reliability performances, both of these characteristics should, in theory, reduce strongly the recurring costs of the spacecrafts.

Such a concept is reinforced by advances in technology in the fields of performances and miniaturisation : these advances are made possible by the research and development programmes led by DARPA and SDIO,these programmes are also based on high technologies developped out of the spatial area. For example :

- digital computers
- mass memory
- ASIC with availability of dedicated algorithmes for digital processing
- silicium and Ga As microelectronics facilities where heavy investments for civilian or military airborne applications give new opportunities for space applications such as : optical detectors, radiofrequency MIMIC which are usuful to design phase array antennas for communication satellites or synthetic aperture radar instruments.
- etc ...

The potential advantages of such a "Smart small cheap lightsats" concept are very attractive for the users, this is particulary true for tactical applications in low earth orbit :

- repetitivity of the passes
- flexibility in the choice of the orbit pa·ameters
- share of risks of failure of each spacecraft and lower vulnerability of the overall system
- flexibility in the way to use the system. In orbit means are fitted with the level of crisis with a continuum from str^tegic use to tactical use
- dedicated satellite for a given mission
- satellite programing and use decentralized to on the field theater users
- etc... (list not exhaustive)

At this point we will not discuss if such a concept is technically relevant in fulfiling the requirements of a given set of missions.

However, with the payload beeing shared between a large number of small satellites, we can assume that the total in orbit mass of these satellites to fulfil a mission would be obviously larger than the total mass when the classical approach using heavy satellites is followed : grouping different payloads or a single satellite always saves mass.

As a consequence, a "Smart small cheap lightsats" approach using satellites with an in orbit life time 5 to 10 times shorter than in a classical approach leads to place in orbit a total mass more than ten times heavier than the total mass necessary to fulfil the same mission on a big satellite.

For a mission on a given period of time Fig. 1 shows the evolution of the launch cost and satellites costs with respect to the targetted in orbit life time for each satellite of the constellation.

For a constant launch unit cost, the total launch cost (curve 1) is the inverse ratio of the in orbit life time.

The satellites total cost (curve 2) is consistent with the "Smart small cheap lightsats" approach : when the in orbit life time decreases, you save more money from the satellites unit cost decrease than you loose from the increase in the number of required satellites, the net result is that the total cost decreases. (this conclusion is less valid for a very long in orbit life time and is no more valid for a very short one).

With a launch unit cost of the same order of magnitude than each satellite unit cost (curves in continuous line) the overall cost (curve 3) decreases when life in orbit increases. At evidence the optimum is to improve the reliability and the in orbit life time: it is the present trend.

A "Smart small cheap lightsats" approach would only become pertinent if very cheap launch costs are proposed (curves in dotted line).

A coherent way to decrease the launch costs is to aim for a "Smart small cheap lightlauncher" concept cutting down the launch cost fo. : Kg in orbit by at least ten times.

Unfortunately the cost per Kg in orbit for present or near term heavy launchers is very far from this goal, for small launchers of the PEGASUS class with a cost per Kg 2 to 3 times greater than the one of the ARIANE V/TITAN class heavy launchers it becomes more and more difficult to achieve.

A WISER APPROACH

With the lack of emergence of very cheap launchers, constellations of numerous cheap lightsats with short in orbit life time are neither for the short term nor for the medium term.

To open the way to new missions -especially in the tactical area - the only driving force of interest is, for a given mission, a moderate overall cost, consistent with the performances offered for a set of sufficiently high priority objectives.

In other words, the today raising of technological breakthrough in performance and miniaturisation will be used in two ways to :

1 - enhance again the present performances of large satellites which will remain heavy and sophisticated. This is typically the case in the reconnaissance field where users put continously more stringent requirements.

2 - open the door to new missions, able to satisfy economically new needs through the use of spatial means. New missions types will be offered while reducing the mass, the size and the overall cost of satellite in orbit. This appears especially pertinent for tactical missions.

CONSEQUENCES FOR THE BUS DESIGN

Due to the reduced number of satellites for a given application, the reduction of the spacecraft recurrent unit cost is no more the key element to reduce the overall cost.The cost reduction has to be more focussed on its non reccuring part than on the recurring one.

As a consequence, mass production savings will not be gained from only one mission, but will have to be gained from a multimission approach.

Assuming that the small satellites potential applications are not aimed at replacing the present heavy and complex satellites systems, the overall performances required from the spacecraft BUS would be moderate. This is very convenient with a modular multimission BUS concept based on :

- standard payload interfaces
- oversized ressources for each module (for ease the sake of a good standardization)
- flexible accomodation of the different modules on the BUS allowing on request modules addition

Fig. 2 shows the modular multimission BUS concept designed jointly by MATRA MARCONI SPACE and its north american partner FAIRCHILD.

In order to reach some non recurrent cost mass production savings through several applications, it is possible to add to this modular approach a new development plan concept based on concurrent engineering.

Fig 3 shows the logic of SYSTEMA . SYSTEMA is a concurrent engineering satellite design tool developped by MATRA MARCONI SPACE. At the center a common data base contains the current configuration of the satellite beeing designed. At the periphery specific tools for each main areas of engineering design are shown. They can be concurrently call on demand by the different experts working in parallel for trade off or validation on different fields : electrical, thermal, mechanical , orbit environment, attitude and orbit control, dynamic , propulsion, payload accomodation, mission, satellite programmation, etc ...

Presently, SYSTEMA is used primarily in the first part of the development process (fig 4), from phase A proposals, mission and satellite layout trade-offs to the end of phase B with detailed satellite design optimisation and up to proposals for a phase C activity.

If additional risks caused by eventual flaws or bugs in numerical simulations are accepted - these risks are assumed to be affordable for small tactical satellites with not very sophisticated missions -, phase C validation by means of such concurrent engineering only tools would save not only the major portion of the environmental tests costs (thermal, vacuum, sun, vibrations) but also would reduce the schedule induced costs.

Such an approach has demonstrated its efficiency through the S80T programme (fig 5). S80T is a microsatellite launched this summer as a companion of TOPEX - POSEIDON SPACECRAFT on ARIANE. S80T has benefited from an intensive use of SYSTEMA from the first stage of design to the delivery at the launch site. A programme conducted in a very short time for a very competitive cost.

CONCLUSION

The approch presented above is far from a revolution. It is only an evolution.

To be efficient such concurrent engineering tools need to be validated and calibrated by reference to the data gathered during previous development tests or during in orbit monitoring of operational spacecrafts. In that sense such an approach capitalises the whole experience acquired by MATRA MARCONI SPACE through previous or current low earth orbit programmes : SPOT 1,2,3 - ERS 1 and 2, HELIOS - SPOT 4 and Polar Platform.

The SYSTEMA use built-in flexibility gives the opportunity to quickly update a current design and thus allows to follow evolutions of technologies or accomodate a newly available equipment.

SYSTEMA allows a short reaction time and a quick accomodation on the satellite as soon as emerging requirements are formulated by the users.

(Illustrations for this Section appear immediately after the French translation.)

DES GROS SATELLITES EN NOMBRE LIMITE...

Avant d'aborder la façon de concevoir une plate-forme multimission pour des applications tactiques en orbite basse, il est utile d'analyser de plus près la notion de satellites tactiques qui est souvent associée aujourd'hui à l'idée de petits satellites pas chers.

Depuis 30 ans on constate que la taille, la masse, la complexité et donc le coût des satellites croissent. Les programmes spatiaux et plus particulièrement dans le domaine de la défense tendent à utiliser des satellites lourds et coûteux en nombre jugé nécessairement trop limite par des utilisateurs qui doivent gérer en permanence des conflits de priorité : conflits budgétaires pour décider quel programme mener en priorité mais aussi conflits de priorité en utilisation opérationnelle pour programmer l'emploi des moyens.

Dans le contexte actuel des compre sions budgétaires on voit ainsi renaître le syndrome bien connu 1 AVION 1 CHAR 1 SATELLITE comme seule dotation à terme pour la défense.

Cette limitation du nombre de satellites est la conséquence d'une spirale inflationniste que l'on peut décrire en trois étapes :

a - augmentation du coût unitaire

b - à budget constant diminution du nombre d'exemplaires opérationnels

c - à faible nombre d'exemplaires optimisation de la mission :

. recherche d'une grande durée de vie et d'une haute fiabilité

. regroupement de plusieurs missions sur une seul satellite

. complexification technique

. méthodes de développement conservatrices à base de vérifications complexes et minutieuses

. refus de prendre des risques devant l'enjeu que représente le coût unitaire de chaque satellite

ce qui entraîne l'augmentation du coût unitaire et nous ramène au point "a "... etc

... AUX CONSTELLATIONS DE PETITS SATELLITES BON MARCHE

Pour sortir de cette spirale certains proposent une approche "Smart sm... cheap lightstats" dont la caractéristique essentielle est de diminuer drastiquement le coût unitaire du satellite en orbite, ce coût comprenant le coût récurrent du satellite, le coût du lancement, le coût du maintien à poste par le centre de contrôle.

Cette approche est cohérente avec une architecture système basée sur des constellations de nombreux satellites dont la durée de vie en orbite est limitée et dont la fiabilité peut être plus faible (droit à la panne), ce qui en théorie permet de réduire fortement les coûts récurrents des satellites.

Une telle approche se trouve confortée par les avancées technologiques en performances et miniaturisation qui alimentent aujourd'hui le spatial au travers des programmes de Recherche et Développement menés par la DARPA ou la SDIO, mais également par synergie avec la haute technologie développée pour des applications autres que spatiales.

Citons :

- le domaine du calcul digital

- les mémoires de masse

- les ASIC de traitements numériques avec la disponibilité d'algorithmes spécialisés

- le domaine des fonderies silicium ou As Ga où des investissements très lourds rentabilisés par des applications civiles ou par l'avionique militaire offrent au domaine spatial soit des détecteurs optiques soit les MIMIC nécessaires pour les antennes réseaux des satellites de communication et les radars à ouverture synthètique

- etc..

Les avantages potentiels de cette approche "Smart small cheap lightsats"sont alléchants pour l'utilisateur, particulièrement dans le domaine tactique sur des orbites basses à défilement :

- répétitivité des passages

- souplesse de choix des paramètres des orbites

- répartition des risques unitaires de perte de chaque satellite et donc moindre vulnérabilité

- souplesse d'emploi : on ajuste ses moyens en orbite à l'état de crise, ce qui permet un passage continu de l'emploi stratégique à l'emploi tactique

- spécialisation des satellites par mission

- décentralisation de la programmation et de l'utilisation du satellite sur le terrain

- etc... cette liste n'étant pas exhaustive.

Sans discuter ici la pertinence technique de cette approche pour satisfaire les performances des diverses missions, on peut prédire que le fait de répartir la charge utile sur un grand nombre de satellites fera que la masse totale en orbite des satellites opérationnels à un moment donné sera certainement supérieure à celle de l'approche classique par gros satellite. On a toujours une prime en masse pour le regroupement des charges utiles.

Avec des durées de vie de 5 à 10 fois plus courtes il faudra donc s'attendre avec une approche de type "Smart small cheap lightsats" à lancer une masse globale plus de 10 fois supérieure pour remplir une mission sur une durée déterminée

Pour une mission de durée donnée, on a représenté (figure 1 -courbe 1) l'évolution en fonction de la durée de vie de chaque satellite du coût de l'ensemble des lancements. La courbe 2 présente également l'évolution du coût de l'ensemble des satellites (coûts récurrents).

A coût unitaire constant, le côut total des lancements (courbe 1) est inversement proportionnel à la durée de vie.

L'évolution du côut total des satellites (courbe 2) reflète l'effet escompté de l'approche "Smart small cheap lightsats" : quand la durée de vie diminue, le gain de coût unitaire compense largement l'augmentation du nombre de satellites nécessaires et le coût global decroît (cet effet n'étant plus vrai pour des durées de vie très très courte et s'atténuant pour des durées de vie longues).

Pour des coûts de lancement de l'ordre de grandeur des coûts satellites (courbes en traits pleins) le coût total (courbe 3) décroît au fur et à mesure que la durée de vie s'allonge. L'optimum est à l'évidence d'améliorer la fiabilité et la durée de vie : c'est la tendance actuelle.

Pour que l'approche "Smart small cheap lightsats" devienne efficace, il faudrait que les coûts unitaires de lancement deviennent beaucoup plus faibles (courbes en pointillé).

Pour réduire de façon cohérente les coûts de lancement il faudrait donc une approche "Smart small cheap lightlaunchers" reduisant au moins par 10 le coût du kilogramme lancé.

Malheureusement les coûts du kilogramme en orbite pour les lanceurs actuels ou en projet sont très loin de cet objectif et on s'en éloigne d'autant plus que l'on cherche à utiliser des petits lanceurs de la classe PEGASUS qui affichent un coût au kilo de 2 à 3 fois supérieur à celui des très gros lanceurs de la classe ARIANE V ou TITAN.

UNE APPROCHE PLUS RAISONNABLE

Faute de voir émerger des lanceurs à très bas coût, les constellations de nombreux petits satellites pas chers à faible durée de vie ne sont ni pour le court ni pour le moyen terme.

Le seul critère réellement dimentionnant pour voir émerger de nouvelles missions et *particulièrement dans le domaine tactique* sera plus certainement d'avoir un coût global de possession raisonnable pour des performances jugées suffisamment prioritaires

Dit encore autrement, les avancées technologiques en performance et miniaturisation que l'on voit poindre aujourd'hui poussent dans deux voies :

- Améliorer encore les performances actuelles des gros satellites qui resteront lourds et complexes. C'est typiquement le domaine où les besoins exprimés par les utilisateurs vont toujours croissant. Par exemmle la reconnaissance.

- Ouvrir la voie vers de nouvelles missions pour satisfaire économiquement de nouveaux besoins au moyen de systèmes spatiaux, en offrant des performances nouvelles et en réduisant la masse, la taille et donc le coût de possession en orbite d'un satellite. Ceci est particulièrement pertinent pour des applications tactiques.

CONSEQUENCES SUR LA CONCEPTION DES PLATES-FORMES

Compte tenu du nombre nécessaicment réduit de satellites pour une application, la réduction du coût unitaire récurrent n'est plus l'élément clef pour réduire le coût global. L'effort de réduction des coûts doit porter plus sur le non récurrent que sur le récurrent.

En corollaire, l'effet de série n'étant plus suffisant sur une seule mission, les réductions de coût liées aux effets d'échelle doivent s'obtenir en jouant sur l'aspect multimission.

Au plan des performances, dans la mesure où les applications envisagées sur petits satellites n'auront pas la prétention de remplacer les systèmes actuels utilisant des satellites lourds et complexes, l'enveloppe des performances demandées à la plate-forme doit être plus modérée. Ceci rend particulièrement pertinente une approche modulaire mettant en oeuvre :

- des interfaces standards avec la charge utile
- une surabondance des ressources pour chaque module autorisant une bonne standardisation
- un aménagement souple favorisant l'ajout de modules complémentaires, à la demande.

Cette approche modulaire est représentée typiquement par les concepts de BUS MULTIMISSION étudiés conjointement par MATRA MARCONI SPACE et notre partenaire nord américain FAIRCHILD (figure 2).

Toujours dans le but de rechercher un effet d'échelle entre plusieurs applications pour réduire les coûts non récurrents, il y a lieu d'adjoindre à cette approche modulaire une nouvelle conception du plan de développement basée sur une ingenierie intégrée.

La figure 3 présente la logique de l'outil SYSTEMA utilisé à MATRA MARCONI SPACE en ingenierie satellite.

Autour d'une base de donnée décrivant la configuration courante du satellite étudié, un certain nombre d'outils spécifiques permettent de mener les études d'ingenierie de façon intégrée dans les grands domaines classiques dans la conception de satellites : analyses électriques, thermiques, mécaniques , d'environnement en orbite, contrôles d'attitude et dynamique, propulsion, implantation des charges utiles, analyse de mission, programmation, etc...

Aujourd'hui, figure 4, ces types d'outils d'ingenierie intégrée sont utilisés principalement dans la première partie d'un développement depuis la proposition, les trade off mission, la phase A , les compromis d'aménagement jusqu'à la phase B et les optimisations fines de la configuration.

Pour réduire encore les coûts non récurrents, et à condition d'admettre les risques complémentaires d'imperfection d'une simulation numérique - risques acceptables pour des applications pas trop complexes de petits satellites tactiques - la validation par simulation de la configuration en utilisant l'outil en phase C devrait permettre de réduire encore très fortement non seulement le coût des essais globaux (vide-soleil, mécanique, dynamique...) mais surtout les temps de développement.

Cette approche a déjà été testée avec succès au travers de S80T (figure 5) microsatellite lancé à l'été 92 en compagnon du satellite TOPEX POSEIDON par ARIANE. L'utilisation intensive de SYSTEMA depuis la phase de conception jusqu'à la fin du développement a permis de mener ce programme à bon terme en très peu de temps et pour un coût très attractif.

CONCLUSION

L'approche présentée ici est loin d'être une révolution, c'est seulement une évolution .

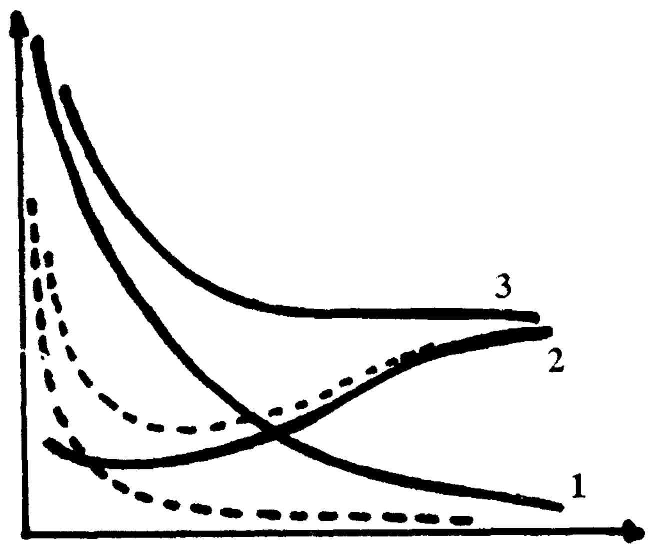
Elle capitalise au travers des outils d'ingenierie intégrée tout l'acquit des plates-formes multimission développées par MMS au travers des programmes SPOT 1, 2, 3, ERS 1 et 2 , HELIOS - SPOT 4 et plate-forme polaire.

Ces outils d'ingenierie intégrée ne prennent en effet toute leur efficacité qu'à l'aune des expériences acquises et des analyses menées lors des développements ou après expertise des comportements en vol.

Une telle approche, par sa souplesse de mise en oeuvre autorise également une conception ouverte à l'évolution technologique et à la prise en compte rapide des nouveautés.

Elle permet de réagir au plus vite aux nouveaux besoins (ou à leur adaptation rapide) dès qu'ils sont exprimés par les utilisateurs.

1 - CONSTELLATION LAUNCHES COST
2 - CONSTELLATION SATELLITES COST
3 - GLOBAL COST

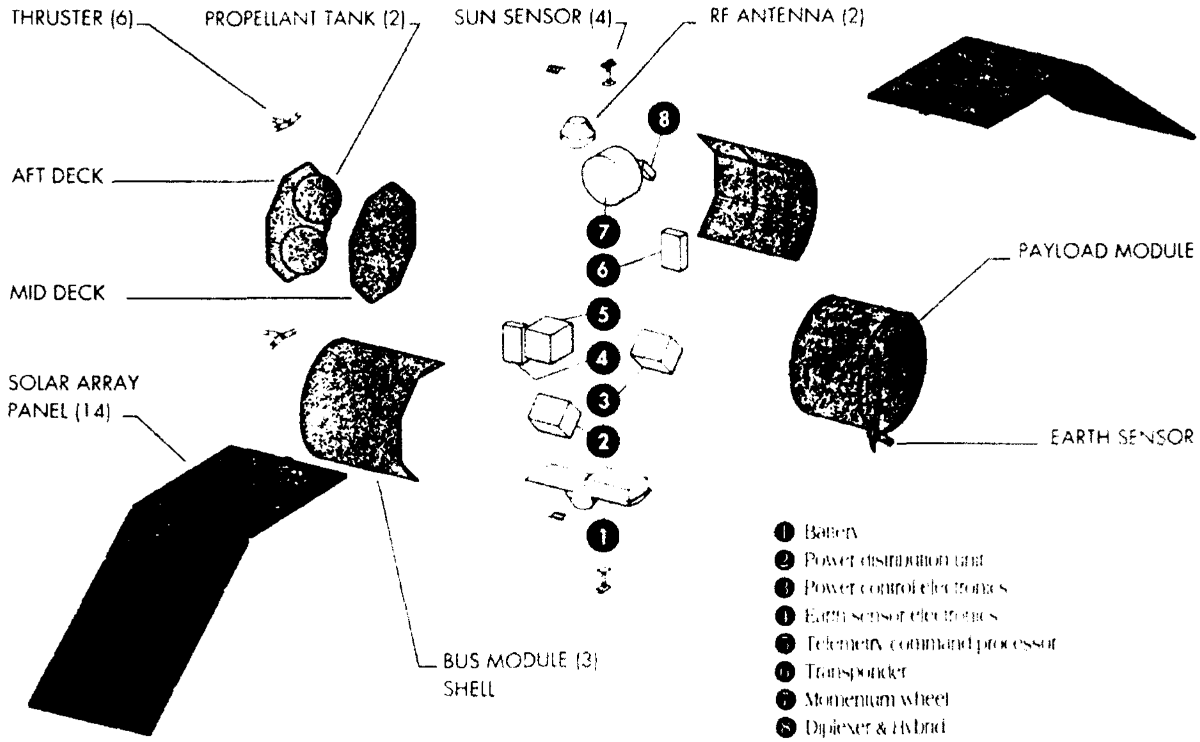


IN ORBIT LIFE TIME
FOR EACH SATELLITE OF
THE CONSTELLATION

GLOBAL COST OF THE CONSTELLATION FOR A MISSION
ON A GIVEN PERIOD OF TIME

FIGURE 1

SPACECRAFT FEATURES



Class II multimission spacecraft

CAPABILITY:

Mission life :	3 to 4 years
Orbit altitudes :	200 - 1000 km
Payload available mass :	up to 200 kg
Payload available power :	100 W mean, 200 W peak
Battery capacity :	9 or 18 Ah
Voltage :	28 +/- 4 V
Attitude pointing accuracy :	+/- 0.15 deg
Data storage on board :	5 Gigabits
Data transmission :	2 to 8 Kbits/sec
Encryption :	optional

SATELLITE BUDGETS:

	Mass	Power
Payload :	62 Kg	110 Watt
Platform :	180 Kg	60 Watt
Hydrazine : (4 years mission)	8 Kg	
Total :	**250 Kg**	**170 Watt**

The FAIRCHILD/MATRA MARCONI SPACE spacecraft is a light weight, low-cost structure designed to accomodate a wide range of scientific and operational payloads . It provides a simple, efficient power system with a super-NiCd battery and a modular solar array that can be adapted to various orbit geometries and payload demands . The central data processor unit includes an embedded solid-state recorder and generous margin for growth . The attitude determination and control system and the propulsion system are both designed for flexibility and long life .

FIGURE 2

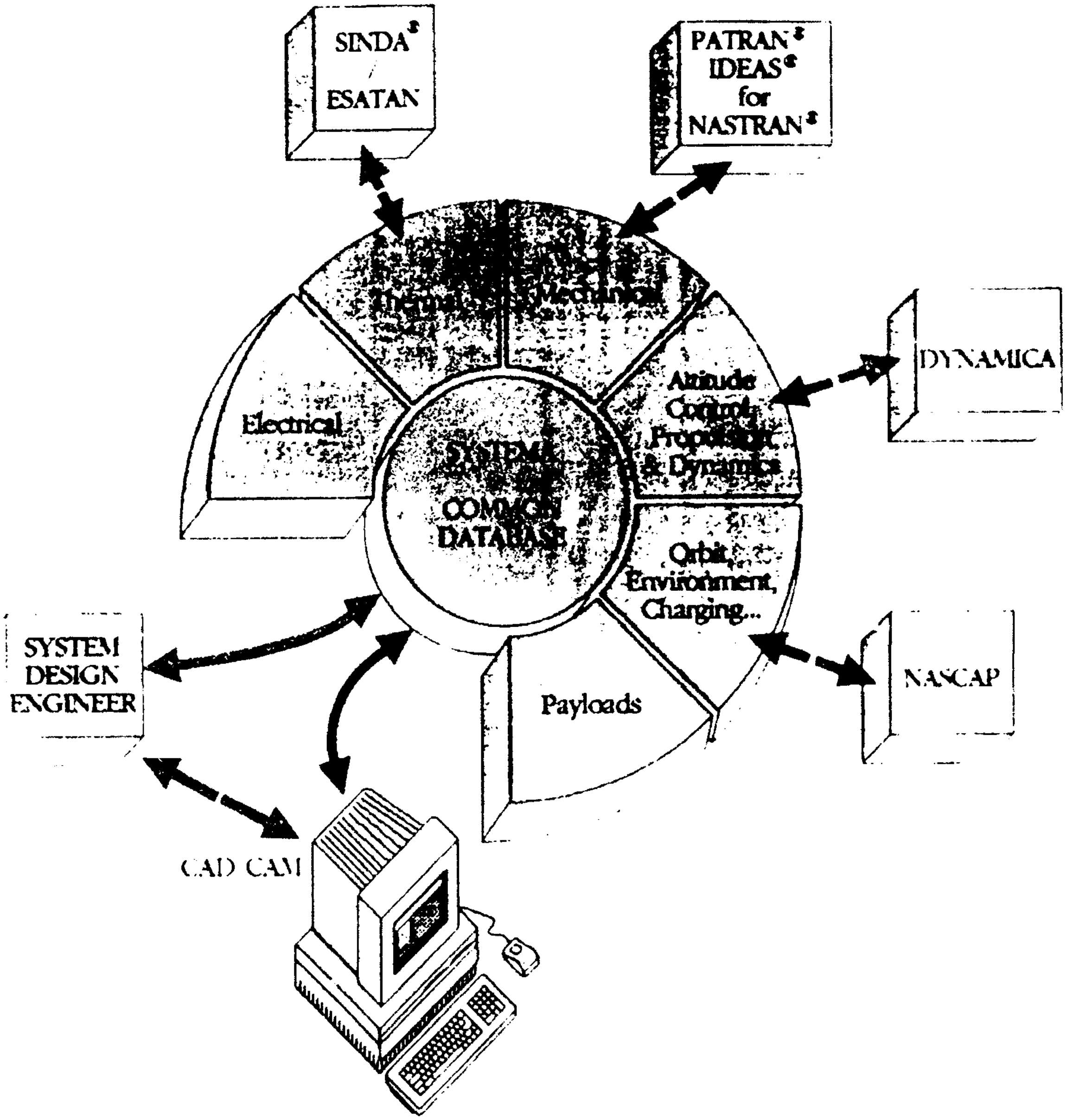

SINDA /
ESATAN
PATRAN
IDEAS
for
NASTRAN
Thermal
Mechanical
Electrical
SYSTEM
COMMON
DATABASE
Attitude
Control,
Propulsion
& Dynamics
DYNAMICA
Orbit,
Environment,
Charging...
NASCAP
Payloads
SYSTEM
DESIGN
ENGINEER
CAD CAM

FIGURE 3

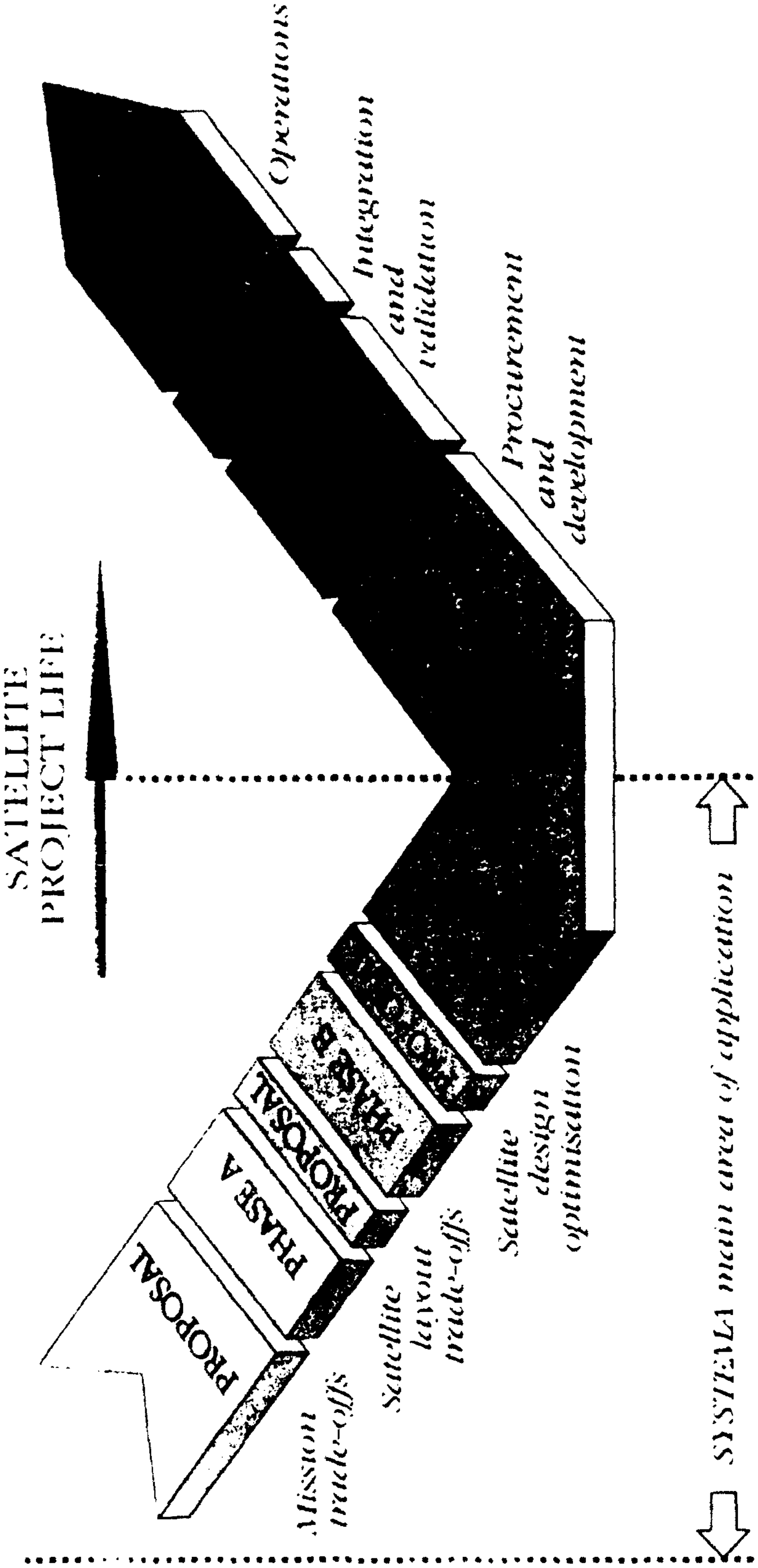

SATELLITE PROJECT LIFE
PROPOSAL
PHASE A
PROPOSAL
PHASE B
Mission trade-offs
Satellite layout trade-offs
Satellite design optimisation
Procurement and development
Integration and validation
Operations
SYSTEMA main area of application

FIGURE 4

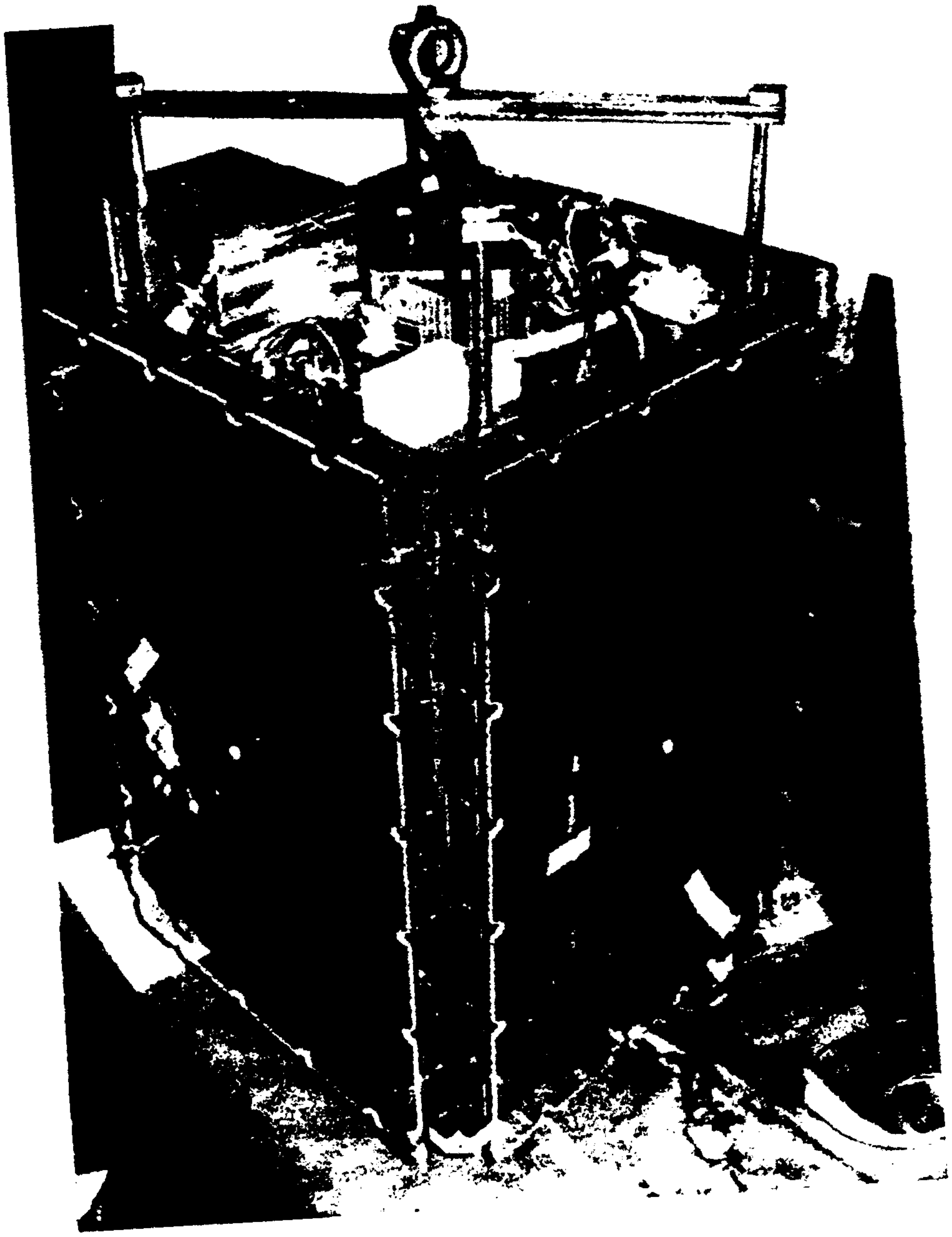


FIGURE 5

SYSTEME DE NAVIGATION PAR SATELLITES A COUVERTURE EUROPEENNE

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1. SOMMAIRE
L'objet de cette présentation est de proposer un système de navigation fonctionnant suivant le même principe que le système GPS mais qui offrirait un service permanent en Europe à l'aide d'un nombre limité de satellites.

La constellation présentée ici ne nécessite que 4 satellites comprenant un géostationnaire et trois satellites sur des orbites géosynchrones d'inclinaison et d'excentricité faibles. Cette constellation permet de couvrir totalement l'Europe, le Moyen Orient et l'Afrique. En outre, l'utilisation de satellites relativement simples est envisageable. En effet, tous les satellites étant visibles en permanence depuis un même point du Globe, il est possible de laisser au sol l'horloge ultra-stable qui fournit la référence de temps pour les mesures de distance usager-satellite.

Cette constellation apparaissant comme prometteuse, une analyse plus poussée est en cours afin de juger de la faisabilité d'un tel système. En particulier, les problèmes posés par la mise et le maintien à poste d'une telle constellation ont d'ores et déjà été analysés. On présente ici plusieurs solutions envisageables pour la mise à poste, avec lancements uniques ou multiples, à partir d'orbites de transfert standard ou non. Afin de fournir une première évaluation du coût de maintien à poste, l'influence des principales perturbations d'orbite sur le service fourni a été analysée. Quelques exemples de maitien à poste sont également fournis.

2. INTRODUCTION
Les performances du système américain de navigation par satellites GPS (Global Positioning System) sont désormais bien connues et ses applications potentielles particulièrement nombreuses tant dans le domaine civil que militaire. Bien entendu, l'obtention de ces performances (couverture mondiale quasi permanente) nécessite la mise en oeuvre d'un nombre important de satellites complexes.

Il nous a donc paru intéressant d'examiner s'il est possible de concevoir des systèmes de navigation fonctionnant suivant le même principe que GPS, mais avec un nombre de satellites beaucoup plus réduit. Le service offert ne diffèrerait de celui fourni par GPS que par l'étendue de la zone géographique couverte par ces systèmes. Nous nous sommes plus particulièrement intéressés à des constellations offrant une couverture de l'Europe.

Les résultats d'une première étude ont montré qu'il est possible d'assurer un service permanent de navigation sur la majeure partie de l'Europe, le bassin méditerranéen et l'Afrique, avec une constellation de 4 satellites, nombre minimal requis pour une navigation de type GPS.

Les résultats présentés dans cette communication concernent une première analyse de la faisabilité d'un tel système, et en particulier la capacité à déployer et maintenir en service la constellation de satellites utilisée.

Tout d'abord, nous rappellerons brièvement le principe de fonctionnement du système de navigation GPS. Puis nous présenterons les principales caractéristiques de la constellation à 4 satellites retenue pour cette étude. Les deux chapitres suivants sont consacrés respectivement à la mise à poste et au maintien à poste de cette constellation.

3. FONCTIONNEMENT DU SYSTEME GPS
Le système américain de navigation par satellites GPS (Global Positioning System) permet à tout usager, disposant du récepteur adéquat, de déterminer presque instantanément sa position, sa vitesse et l'heure locale. Ces calculs sont effectués avec une précision inégalée jusqu'à présent par les autres systèmes de navigation. Le principe de navigation utilisé, par triangulation géométrique redondante, impose à tout usager du système d'être en visibilité simultanée d'au moins 4 satellites de la constellation. Dans sa version complète, celle-ci est composée de 24 satellites évoluant sur des orbites circulaires inclinées d'une période de 12 heures.

Rappelons brièvement le principe de fonctionnement du système. Chaque satellite de la constellation GPS émet en permanence un signal vers la Terre. Ce signal codé contient la date exacte de départ du signal, qui est déterminée par une horloge atomique embarquée, et des éphémérides qui permettent de connaître très précisément la position et la vitesse du satellite à cet instant. Ainsi un récepteur approprié, doté d'une horloge, peut déterminer la distance qui le sépare du satellite à partir de l'écart entre les dates de départ et d'arrivée du signal. La vitesse radiale du satellite par rapport au récepteur peut également être déterminée par mesure de l'effet Doppler sur le signal. En principe, avec trois satellites en visibilité sous un site minimal de 5°, l'usager pourrait déterminer par triangulation sa position et sa vitesse.

Un système de contrôle au sol permet de maintenir une excellente synchronisation des horloges de grande précision des satellites. Par contre, l'horloge de l'usager est beaucoup moins stable. En l'absence de recalages fréquents, le temps qu'elle indique finit par différer notablement du temps de référence GPS, ce qui introduit une forte erreur dans la détermination des distances usager-satellites: celles-ci deviennent alors inexploitables.

L'erreur sur la distance est la même pour tous les satellites observés au même instant. Elle peut donc être estimée à partir d'une mesure sur un satellite supplémentaire. Ainsi, si un usager du système fait des mesures sur 4 satellites à un instant donné, il peut déterminer à la fois son biais d'horloge (par rapport au temps GPS) ses trois coordonnées géographiques, et éventuellement les trois composantes de sa vitesse. Par la suite, il peut déterminer sa position et sa vitesse avec 3 satellites seulement pendant la courte période durant laquelle son biais d'horloge reste quasi constant.

En définitive, on considérera qu'un usager doit être en visibilité simultanée d'au moins 4 satellites pour pouvoir calculer sa position et sa vitesse instantanées. C'est une condition stricte qui assure une exploitation correcte des signaux émis par les satellites, quelle que soit la qualité de l'horloge de l'usager.

Un usager est capable de déterminer sa position exacte s'il connaît la distance géométrique (même biaisée) qui le sépare de la position diffusée de chaque satellite. La mesure de cette distance est entachée d'une erreur résultant des imperfections du traitement du signal, de la propagation atmosphérique et des éphémérides des satellites. Si les erreurs de mesures pour les différents satellites utilisés sont indépendantes et de même niveau, la précision de localisation 3D, caractérisée par l'écart type de l'erreur d'estimation de la position, est égale au produit de l'écart type des mesures et du PDOP[1] (Position Dilution of Precision). Le PDOP est une quantité sans dimension ne dépendant que de la géométrie relative usager-satellites.

Le PDOP permet d'étudier la précision de localisation de toute constellation de navigation du type GPS indépendamment de la précision des mesures. Ainsi on considère usuellement que le service de navigation est assuré à un instant et en un point donnés si 4 satellites sont visibles et si le PDOP, calculé en ce point, est inférieur à 6.

4. CONSTELLATION OPTIMALE A 4 SATELLITES

Dans une première étude[2], nous avons recherché des constellations à faible nombre de satellites (entre 4 et 9) permettant d'assurer un service de navigation du type GPS sur une région limitée de la Terre.

A nombre de satellites fixé, nous avons cherché à maximiser l'aire de la portion de la surface terrestre à l'intérieur de laquelle le PDOP est en permanence inférieur à 6. Ce critère présente de très nombreux maxima locaux, c'est pourquoi nous avons utilisé une méthode d'optimisation globale dont le principe est une recherche aléatoire adaptive: Adaptive Random Search (ARS). Il s'agit d'une méthode initialement proposée par Bekey et Masri en 1983.[3][4]. Elle ne nécessite que le seul calcul critère et non celui de son gradient.

Les paramètres optimisés sont, pour chaque satellite, 5 des 6 paramètres indépendants définissant son orbite: excentricité, inclinaison, argument du périgée, ascension droite du noeud ascendant et anomalie moyenne. En effet, nous avons choisi pour demi grand axe celui de l'orbite géostationnaire (42 164 km). De la sorte, la zone survolée par chaque satellite est entièrement cantonnée à l'intérieur d'une fraction de la surface de la Terre, ce qui est particulièrement intéressant pour obtenir un service de navigation régional. En définitive, le nombre de paramètres optimisés est compris entre 20 et 45.

Parmi les constellations obtenues à l'issue de cette étude, il en est une particulièrement intéressante qui permet de réaliser une couverture permanente de l'Europe et de l'Afrique avec seulement 4 satellites (voir figure 1), ce qui est le nombre minimal requis. Cette constellation est constituée d'un satellite géostationnaire et de trois satellites situés sur des orbites géosynchrones inclinées ($i=18,3°$) et faiblement excentriques ($e=0,19$).
La couverture permanente de cette constellation représente 15% de la surface terrestre.

Les quatre satellites étant constamment visibles de tout point de la zone couverte, cela permet d'envisager un système de navigation simplifié utilisant une horloge atomique au sol, les satellites étant alors munis d'un simple répéteur.

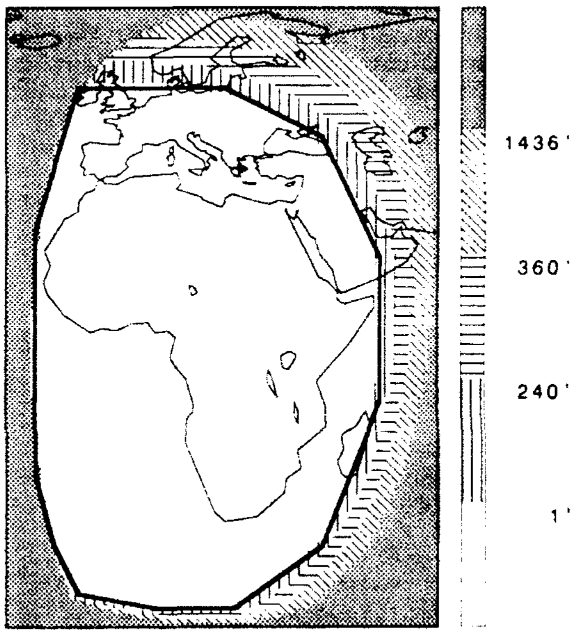

Figure 1: Couverture nominale de la constellation
Indisponibilité journalière (mn)

5. MISE A POSTE

La mise à poste d'un satellite est l'opération qui permet de le placer sur son orbite de service à partir de la surface de la Terre. Elle est généralement réalisée en deux étapes, surtout dans le cas d'orbites élevées qui nous concerne dans cette étude:

- un lanceur injecte le satellite sur une orbite intermédiaire qui diffère de l'orbite finale souhaitée.
- le satellite effectue un transfert de l'orbite intermédiaire à l'orbite finale au moyen d'un moteur qui lui est propre.

Dans le cas d'une constellation de satellites, on peut utiliser des lancements multiples afin de réduire le coût global de l'opération: la première étape est faite simultanément pour deux satellites ou plus.

Le transfert d'un satellite à partir de l'orbite intermédiaire a toujours été déterminé dans cette étude de façon à minimiser la quantité d'ergols nécessaire à l'opération, à masse finale du satellite fixée. Aucune contrainte n'a été prise en compte. En particulier, la poussée du moteur est supposée librement orientable.

Les manoeuvres ont été modélisées par des impulsions. Dans ces conditions, la masse d'ergols m_e consommée par un transfert s'exprime simplement par:

$$m_e = m_f \left[e^{\frac{\Delta V}{V_e}} - 1 \right]$$

avec:

m_f : masse finale du satellite; c'est la masse à l'issue du transfert, masse du moteur comprise,
ΔV: somme des variations de vitesse du satellite imprimées par les diverses impulsions du transfert,
V_e : vitesse d'éjection du moteur.

Minimiser la quantité d'ergols consommée est donc équivalent à minimiser la somme des variations de vitesse ΔV. Ce dernier critère présente l'intérêt d'être indépendant des performances du moteur (V_e) et de la masse du satellite. C'est donc lui qui a effectivement été employé.

Dans cette étude, chaque transfert est supposé réalisé au moyen de 2 impulsions, ce qui est le nombre minimal requis pour un transfert quelconque (entre des orbites non sécantes).

5.1 Mise à poste du satellite géostationnaire

On suppose que le satellite géostationnaire sera mis à poste par la procédure standard suivante:

- un lanceur ARIANE injecte le satellite sur une orbite GTO (Geostationary Transfer Orbit) dont les paramètres orbitaux sont approximativement les suivants:

Demi grand axe:	24 372 km
Excentricité:	0.73
Inclinaison:	8°
Argument du périgée:	180°

Ascension droite du noeud ascendant: **non fixée** (elle est fonction de l'instant de lancement)

- le satellite est placé sur l'orbite finale au moyen d'une impulsion unique effectuée à l'apogée de l'orbite GTO. Il s'agit de son seul point d'intersection avec l'orbite géostationnaire. Avec les valeurs fournies ci-dessus, le coût du transfert est:

$$\Delta V = 1509 \text{ m/s}$$

Pour une vitesse d'éjection usuelle V_e= 3000 m/s, le transfert nécessite donc une quantité d'ergols égale à:

65% de la masse finale.

5.2 Mise à poste des autres satellites

En ce qui concerne la première étape de la mise à poste, quatre solutions ont été envisagées:

a) les satellites sont injectés initialement sur une orbite GTO (ses caractéristiques sont celles indiquées au paragraphe précédent),

b) ils sont injectés sur une orbite dédiée à leur mise à poste; il s'agit d'une orbite de paramètres orbitaux identiques à ceux d'une GTO, à l'exception de l'inclinaison, égale à l'inclinaison de l'orbite finale,

c) ils sont injectés sur une orbite dédiée à leur mise à poste; les paramètres de l'orbite sont déterminés de façon à ce qu'elle intersecte les 3 orbites finales; chaque transfert est réalisé au moyen d'une impulsion unique effectuée en l'un de ces points.

d) ils sont injectés sur une orbite dédiée à leur mise à poste, dont les paramètres sont totalement optimisés; plus précisément, on optimise conjointement la trajectoire de montée du lanceur et les impulsions permettant de réaliser le transfert.

Pour les deux premières solutions, nous avons étudié deux procédures de mise a poste:

- **chacun des 3 satellites est injecté séparément** par un lanceur propre.

Pour l'un des satellites, on détermine alors le transfert bi-impulsionnel optimal qui permet de passer de l'orbite d'injection à l'orbite finale. En réalité, l'orbite d'injection n'est pas entièrement définie: l'ascension droite Ω_0 de son noeud ascendant peut prendre n'importe quelle valeur selon l'instant de tir. La détermination du transfert optimal doit donc être faite pour toutes les valeurs de Ω_0 comprises entre 0 et 2π. On peut alors tracer la courbe d'évolution du coût du transfert en fonction de Ω_0. On en déduit la valeur optimale de Ω_0 qui conduit au transfert "optimum optimorum".

Pour les deux autres satellites, les courbes se déduisent de la précédente par un simple décalage en Ω_0. Ce décalage est égal à l'écart angulaire qui existe entre le noeud ascendant du premier satellite et celui du satellite considéré. En particulier, le transfert "optimum optimorum" a le même coût pour les 3 satellites. Pour pouvoir l'utiliser effectivement, il suffit d'ajuster pour chaque satellite la valeur de Ω_0 par le biais de l'instant de lancement.

- **les 3 satellites sont injectés simultanément** au moyen d'un lanceur unique.

Il n'existe alors qu'une seule orbite d'injection pour les 3 satellites, caractérisée par un Ω_0 donné. Il n'est donc pas possible d'utiliser le transfert "optimum optimorum" pour chacun d'eux. La consommation d'ergols est par conséquent toujours supérieure ou égale à celle du cas précédent. On bénéficie par contre d'un coût de lancement inférieur.

Pour toute valeur de Ω_0 on peut déduire de la courbe obtenue dans le cas précédent le coût du transfert optimal pour chacun des 3 satellites. Ces 3 coûts permettent de calculer un coût global pour les 3 transferts: nous avons choisi le ΔV du transfert unique qui consommerait une masse égale à la masse totale d'ergols des 3 transferts, pour placer sur orbite la même masse finale (3 fois la masse d'un satellite après mise à poste). La valeur de ce critère dépend de la vitesse d'éjection des moteurs: nous avons choisi la valeur V_e=3000 m/s.

Pour les deux dernières solutions, seul le lancement simultané des 3 satellites a été étudié.

5.2.1 Transferts depuis une GTO

La courbe de la figure 2 représente le coût minimal (ΔV) du transfert bi-impulsionnel optimal depuis une orbite GTO, en fonction de l'ascension droite Ω_0 du noeud ascendant de cette orbite.

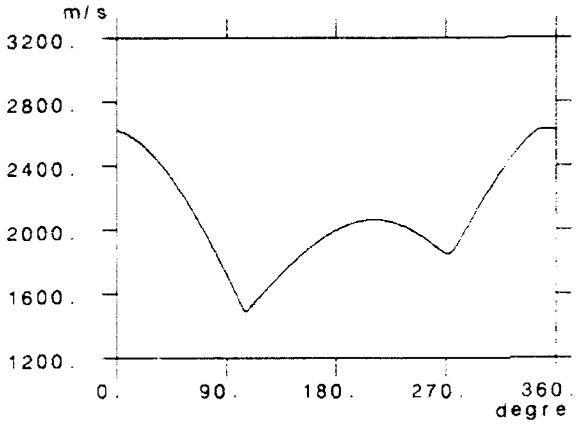

Figure 2: Coût du transfert d'un des satellites en fonction de l'ascension droite du noeud ascendant de l'orbite d'injection (GTO)

La courbe présente deux minima séparés de 165° en Ω_0. Le plus petit est atteint pour Ω_0=106°, soit une valeur proche de celle de l'ascension droite du noeud ascendant de l'orbite finale (102.316°). Le coût correspondant est de:

1484 m/s

1ère impulsion : 15 m/s
2ème impulsion: 1469 m/s

Avec une vitesse d'éjection du moteur de 3000 m/s, la masse d'ergols consommée serait alors de **64%** de la masse finale mise à poste. Ces résultats sont très voisins de ceux du transfert géostationnaire effectué dans les mêmes conditions (1509 m/s, 65%).

Pour pouvoir effectuer chaque transfert pour un coût de 1484 m/s, il faut impérativement utiliser un lanceur pour chaque satellite. L'instant du premier lancement n'a pas d'importance pour la mise à poste. Par contre, les deux lancements suivants doivent être faits à une heure particulière de la journée (qui dépend de la date de l'année).

Si les 3 satellites sont injectés simultanément par un lanceur unique, on évite ces contraintes de temps délicates à respecter. La figure 3 représente le coût des 3 transferts, ainsi que le coût global, en fonction de l'ascension droite Ω_0 du noeud ascendant d'une GTO unique. Il apparaît que le coût global est quasi constant, et qu'il ne permet donc pas de départager les solutions. Par contre, les coûts individuels varient fortement. Une situation particulièrement intéressante apparait pour 3 valeurs de Ω_0: 59°, 179°, 299°. C'est dans ces cas que le plus grand des 3 ΔV est minimal. Les ΔV des 3 transferts sont alors regroupés au maximum et valent:

2160 m/s, 2160 m/s et 1990 m/s

Traduits en masse d'ergols consommée, les résultats deviennent respectivement:

105%, 105% et 94% de la masse finale d'un satellite.

Ces chiffres sont suffisamment voisins pour qu'on puisse envisager de dimensionner de la même façon le système propulsif de chaque satellite.

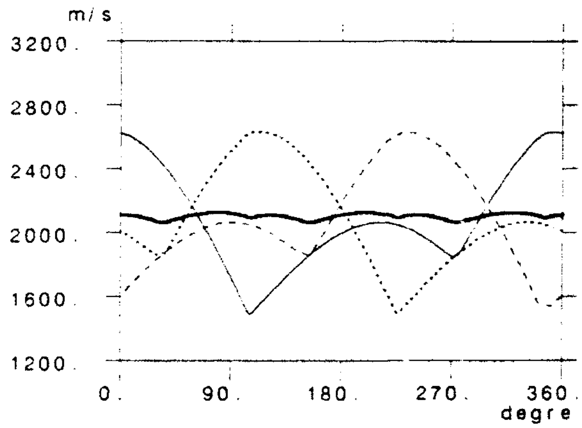

Figure 3: Coût des 3 transferts et coût global en fonction de l'ascension droite du noeud ascendant de l'orbite d'injection (GTO)

La valeur de Ω_0 dépend de l'instant de tir du lanceur. On pourrait donc penser que celui-ci doit être choisi convenablement pour que les transferts puissent être réalisés dans les conditions particulièrement intéressantes que l'on vient de mentionner. En réalité, ce n'est pas le cas parce que les valeurs absolues de l'ascension droite du noeud ascendant des orbites finales n'ont pas d'importance: seules leurs valeurs relatives sont fixées. Or ces valeurs relatives sont respectées dans toutes les configurations représentées sur la figure 3. Donc dans le cas d'un lancement simultané des 3 satellites, la réalisation des transferts n'impose donc aucune contrainte sur l'instant de tir.

5.2.2 Transferts depuis une GTO modifiée

Tous les résultats relatifs à ce cas sont visualisés sur la figure 4, qui est l'analogue de la figure 3: le ΔV de chaque transfert, ainsi que le ΔV global, sont représentés en fonction de l'ascension droite Ω_0 du noeud ascendant de l'orbite d'injection.

Dans le cas d'un lancement individuel de chaque satellite, le transfert peut être moins coûteux que si l'orbite d'injection est une GTO standard. La courbe relative à chaque transfert présente en effet un minimum unique (au lieu de 2 précédemment), dont le ΔV est:

1292 m/s

ce qui correspond à une masse d'ergols consommée de **54%** de la masse finale.

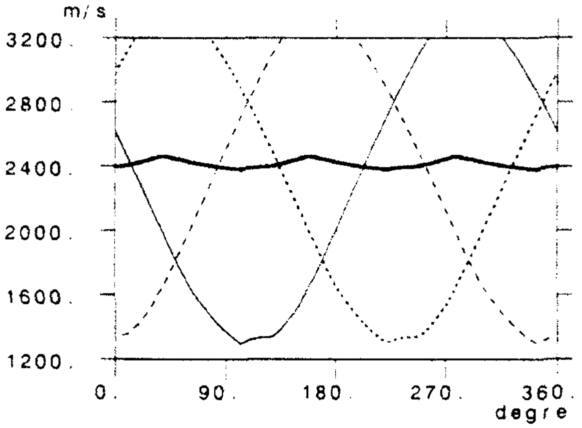

Figure 4: Coût des 3 transferts et coût global en fonction de l'ascension droite du noeud ascendant de l'orbite d'injection (GTO d'inclinaison modifiée)

Si par contre les 3 satellites sont injectés sur une même orbite à l'issue d'un lancement multiple, la situation est plus défavorable que dans le cas d'une injection sur GTO. Il existe 3 solutions optimales, pour Ω_0 = 108°, 228° et 348°. Les coûts correspondants sont:

2797 m/s, 2797 m/s et 1315 m/s

Le ΔV maximal est très nettement supérieur a celui constaté dans le cas d'une injection sur GTO (2797 m/s au lieu de 2160 m/s). En outre, les trois ΔV ne sont pas du tout du même niveau.

En conclusion, l'injection sur GTO modifiée n'est intéressante que dans le cas des lancement séparés. Elle permet de fair une petite économie d'ergols (16%) par rapport à une injection sur GTO standard. Il reste à étudier si une telle orbite est accessible à un lanceur Ariane. La GTO standard est en revanche une orbite d'injection bien supérieure si les 3 satellites sont lancés ensemble.

5.2.3 Transferts mono-impulsionnels depuis une orbite d'injection commune

Nous avons recherché une orbite d'injection interceptant les 3 orbites finales, de façon à pouvoir réaliser chaque transfert au moyen d'une impulsion **unique**. On considère que les satellites sont tous les 3 situés sur cette orbite après injection, ce qui suppose en pratique qu'ils ont été lancés ensemble.

Un balayage exhaustif de toutes les orbites (circulaires ou elliptiques) interceptant les 3 orbites finales a été effectué. Pour chaque solution, nous avons calculé la somme des 3 ΔV des impulsions. En définitive, il est apparu que la solution qui minimise cette somme est une orbite équatoriale de demi grand axe:

42 755 km

Il s'agit donc d'une orbite voisine de l'orbite géostationnaire, dont le demi grand axe est 42 164 km. Toutes les impulsions sont égales et valent chacune:

1124 m/s

En pratique, on pourrait envisager d'injecter dans un premier temps les 3 satellites sur GTO standard, puisque l'orbite optimale identifiée est quasi geostationnaire. Ils seraient ensuite transférés sur celle-ci ensemble ou individuellement, pour un coût ΔV = 1530 m/s. Le coût global pour chaque satellite serait donc de 2654 m/s, soit beaucoup plus que les 2160 m/s nécessaires pour une mise à poste bi-impulsionnelle directe depuis une GTO. Une telle solution présenterait donc peu d'interêt.

5.2.4 Optimisation couplée de la trajectoire du lanceur et des 3 transferts

Nous disposons d'un logiciel capable d'optimiser simultanément la trajectoire de montee d'un lanceur donné, plaçant un ou plusieurs satellites sur orbite d'injection, et les transferts bi-impulsionnels permettant de placer ces satellites sur leur orbites finales[5]. Plusieurs criteres sont susceptibles d'être minimisés. Pour cette étude, nous avons choisi la valeur maximale des ΔV des 3 transferts.

Ce logiciel a été utilisé pour optimiser la mise poste complète des 3 satellites, dans l'hypothese ou on utilise un unique lanceur du type Ariane 5, tire depuis Kourou. Nous avons considéré que la masse finale des satellites en fin de mise a poste est de 500 kg, et que la vitesse d'éjection des moteurs de transfert est de 3000 m/s.

L'orbite d'injection identifiée a l'issue de l'optimisation est une orbite voisine de la GTO standard. Elle est caractérisée par les paramètres suivants:

Demi grand axe:	27 266 km
Excentricité:	0.7299
Inclinaison:	6.45°
Argument du périgée:	172.94°

La figure 5 est l'analogue des figures 3 et 4; elle représente le coût des 3 transferts, ainsi que le coût global, en fonction de l'ascension droite Ω_0 du noeud ascendant de cette orbite.

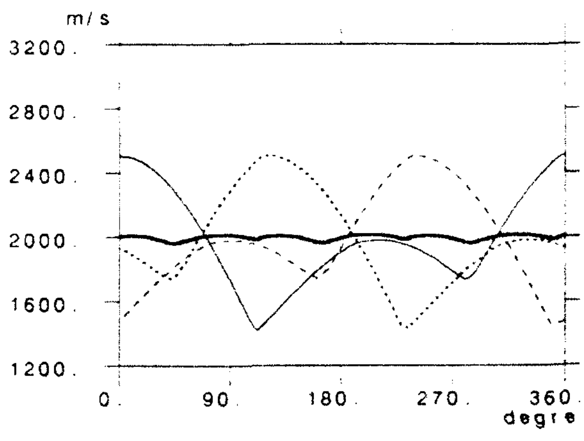

Figure 5: Coût des 3 transferts et coût global en fonction de l'ascension droite du noeud ascendant de l'orbite d'injection (optimisée)

Comme pour la mise à poste à partir d'une injection sur GTO, on constate que la solution la plus intéressante correspond a 3 valeurs de Ω_0. Les ΔV correspondants sont:

2031 m/s, 2031 m/s et 1942 m/s

C'est dans ce cas que la valeur maximale des 3 ΔV est la plus petite. C'est dans ce cas également qu'ils sont les plus voisins les uns des autres. Ces résultats correspondent à des masses d'ergols consommées valant respectivement:

97%, 97% et 91% de la masse finale d'un satellite.

Par rapport à une mise à poste utilisant une GTO standard, on observe une réduction d'environ 8% de la masse d'ergols maximale.

6. MAINTIEN A POSTE

L'analyse de performance qui a permis de proposer la constellation étudiée a été réalisée sans prendre en compte les principales perturbations naturelles agissant sur les satellites de la constellation. La prise en compte de celles-ci est en effet inutile pour analyser la performance du système sur une durée de 24 heures. Cependant, il est clair que le service fourni par le système envisagé n'a d'intérêt que si celui-ci est disponible pendant une durée nominale s'étendant sur plusieurs années. Il convient de vérifier que la constellation proposée est viable vis à vis de ce critère, et qu'en particulier les perturbations subies par les satellites composant la constellation n'imposent pas des manoeuvres de maintien à poste trop fréquentes ou trop coûteuses.

L'étude du maintien à poste a donc été réalisée en deux temps: tout d'abord, les principales perturbations agissant sur les satellites ont été simulées, et l'influence des dégradations d'orbite sur la performance du système a été évaluée. Dans un deuxième temps, pour un cycle de correction d'orbite choisi au vu de l'analyse précédente, des manoeuvres de maintien à poste ont été déterminées et le coût global de maintien à poste a été évalué.

Le satellite géostationnaire n'a pas été pris en compte pour cette analyse pour les raisons suivantes: comme il a été indiqué en introduction, la charge utile de navigation proprement dite sera d'un volume et d'une masse limitée, ce qui autorise son adjonction sur un satellite géostationnaire dévolu à une autre activité principale (télécommunication par exemple). Dans ces conditions, le maintien à poste de ce satellite sera fixé par les exigences de positionnement fin du satellite de télécommunication. Par ailleurs, si l'on envisage d'utiliser un satellite géostationnaire purement dédié à la mission de navigation, l'encombrement de l'orbite géostationnaire imposera à celui-ci une fenêtre de stationnement limitée. Le respect de cette fenêtre nécessitera un maintien à poste d'un niveau comparable à celui d'un satellite géostationnaire de télécommunication, soit environ 50 m/s/an en terme de vitesse caractéristique.

6.1 Usure naturelle des orbites

Compte tenu de la nature des orbites, les perturbations principales qu'il convient de prendre en compte sont les suivantes:

- dissymétries du potentiel gravitationnel terrestre,
- attraction gravitationnelle de la Lune et du Soleil,
- pression de radiation solaire directe.

Le potentiel terrestre a été modélisé en prenant en compte les termes zonaux et tesséraux jusqu'à l'ordre et le degré 8.

Pour le calcul de la pression de radiation, les caractéristiques physiques retenues pour les satellites sont les suivantes:

surface exposée:	3 m^2
masse:	500 kg
coefficient thermo-optique global:	1.2

6.1.1 Intégration du mouvement des satellites

L'analyse de l'usure des orbites sous l'influence des perturbations naturelles a été réalisée en utilisant un premier logiciel développé à l'ONERA qui permet une intégration rapide et fiable du mouvement orbital. Ce logiciel utilise le formalisme des paramètres centrés (ou paramètres moyens) qui permet de supprimer les perturbations à haute fréquence, qui n'ont que peu d'influence sur l'évolution à long terme de l'orbite. Cela autorise ainsi l'utilisation de grands pas d'intégration (typiquement de l'ordre de la période orbitale, soit 24 heures ici), au lieu de quelques minutes si l'on utilise les éléments osculateurs de l'orbite (éléments instantanés). Cette technique permet donc de réaliser rapidement des intégrations de longue durée.

Pour analyser l'évolution de la performance du système sous l'influence des perturbations naturelles, les paramètres orbitaux des trois satellites considérés sont intégrés avec le logiciel précédemment décrit. Périodiquement, on relève la valeur de ces paramètres et on les utilise pour déterminer la nouvelle couverture du système après perturbation d'orbite. En pratique, les calculs de couverture ont été réalisés tous les 7 jours et la durée globale d'analyse consécutive est de 6 mois. La date de départ de l'analyse a été fixée au 1 Janvier 1993. Cette date a une influence directe sur le résultat puisqu'elle fixe la position relative de la Terre et des autres astres perturbateurs. Une analyse plus poussée devra donc être réalisée, pour évaluer notamment l'influence de la date de lancement.

6.1.2 Calcul de la couverture

Afin de pouvoir comparer correctement la couverture de la constellation "perturbée" avec la couverture nominale, il a été procédé de la manière suivante:

- Le contour de la zone de couverture nominale a été modélisé approximativement par un polygone sphérique, que l'on peut observer sur la figure 1.

- Un réseau de points situés à la surface de la Terre a été défini à l'intérieur de ce polygone. Ces points sont répartis sur des parallèles disposés tous les 2° en latitude. Sur chaque parallèle, les points sont équi-répartis en longitude et leur nombre est choisi de telle façon que la densité surfacique de points soit approximativement constante sur l'ensemble du réseau.

- L'analyse de couverture est réalisée en intégrant la constellation sur 24 heures (période de recouvrement de la constellation nominale) et en testant toutes les 5 minutes la valeur du PDOP en chaque point du réseau. On détermine ainsi le pourcentage des points où le service de navigation est accessible en permanence sur la durée d'intégration. Le service est dit accessible à un instant donné si il y a au moins 4 satellites visibles à cet instant, et si le PDOP est inférieur à 6. Compte tenu de la répartition des points du réseau, ce pourcentage est aussi le pourcentage de la surface couverte. Il est exactement de 98,6% pour la constellation nominale. La performance de la constellation dégradée a toujours été chiffrée au moyen de cet indice.

6.1.3 Résultats de l'analyse
La figure 6 représente l'évolution du pourcentage de la surface du polygone couverte par la constellation en fonction du temps (exprimé en semaines).

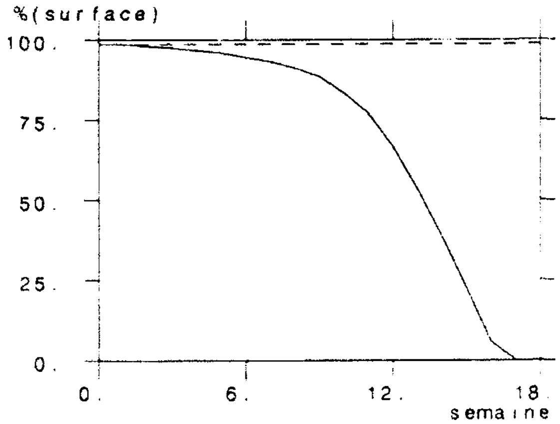

Figure 6: Evolution naturelle du pourcentage de couverture en fonction du temps

Cette figure montre que la couverture diminue lentement et progressivement pendant 6 semaines, puis cette diminution s'accélère pour aboutir à une disparition complète de la couverture après 4 mois environ.

La figure 7 représente la couverture de la constellation à cet instant.

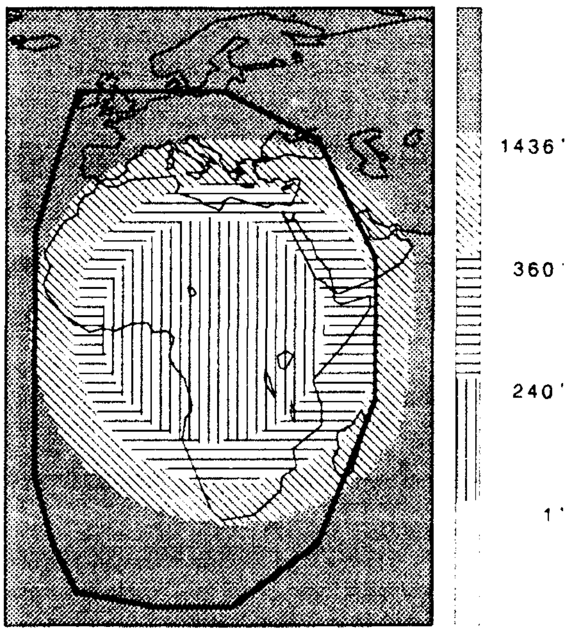

Figure 7: Couverture de la constellation après 4 mois Indisponibilité journalière (mn)

Plus aucune région ne bénéficie d'un service permanent et la zone la plus privilégiée subit des périodes d'indisponibilité journalière pouvant atteindre 4 heures. En outre, les régions Européennes sont très vite privées de service. Il est donc clair qu'un maintien à poste relativement fréquent est nécessaire.

6.2 Stratégie et coût de maintien à poste

6.2.1 Fréquence de maintien à poste
Le premier paramètre à choisir pour évaluer le coût de maintien à poste est la fréquence des corrections. Plusieurs facteurs peuvent influer sur son choix:

- le niveau de dégradation tolérable pour la zone de couverture.
- la durée d'indisponibilité du service: les manoeuvres rendent très certainement le système temporairement inutilisable (la précision de localisation des satellites est alors trop faible pour permettre une navigation de qualité).
- le coût global du maintien à poste: la fréquence des manoeuvres peut dans une certaine mesure influer sur ce coût en cas de non linéarité du niveau des perturbations en fonction du temps.

Pour cette première analyse, nous avons décidé de procéder à un recalage des orbites des satellites toutes les 8 semaines. La figure 8 permet de juger de la dégradation de la performance obtenue après un tel délai sans recalage. On peut constater que le service s'est essentiellement dégradé au Sud, et que l'Europe reste correctement couverte. Le choix des 8 semaines semble constituer un bon compromis: une durée plus longue conduit à une perte de couverture trop importante sur l'Europe, alors qu'une fréquence de manoeuvre plus élevée diminuerait fortement la disponibilité du service.

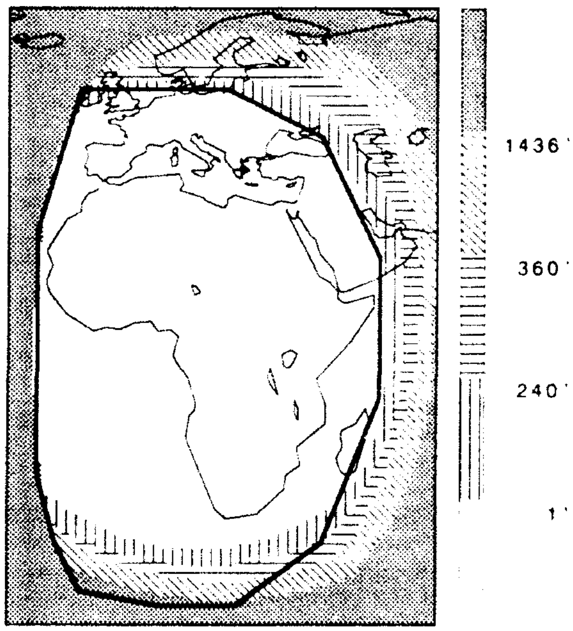

Figure 8: Couverture de la constellation après 8 semaines Indisponibilité journalière (mn)

L'analyse du coût de maintien à poste doit être réalisée en deux temps: tout d'abord, il convient de déterminer quels sont les paramètres qu'il est absolument nécessaire de corriger: il est en effet possible que la sensibilité de certains paramètres sur le service soit suffisamment faible pour que l'on puisse se permettre de ne pas les corriger sur toute la durée du service. Ensuite, le coût des corrections à réaliser doit être calculé. Pour cela, le logiciel d'optimisation de manoeuvres bi-impulsionnelles, déjà utilisé pour l'étude de la mise à poste, a de nouveau été mis en oeuvre.

6.2.2 Manoeuvres de maintien à poste

Une étude exhaustive des possibilités de maintien à poste de la constellation a été réalisée, en testant successivement la nécessité de corriger chacun des paramètres orbitaux. Compte tenu de la volonté de réaliser des satellites simples et peu coûteux, les premières analyses ont eu pour but de vérifier s'il était possible de se contenter de manoeuvres effectuées selon la vitesse du satellite, ou à la rigueur dans le plan de l'orbite. Le but était d'éviter si possible des manoeuvres de modification du plan de l'orbite, qui sont coûteuses et qui imposent une plate-forme plus complexe pour le satellite.

Ces tests ont malheureusement montré que cela n'est pas possible. La figure 9 représente l'évolution de la couverture du système dans l'hypothèse de corrections dans le plan de l'orbite seulement (correction du demi grand axe, de l'excentricité, de l'argument du périgée et de l'anomalie vraie de l'orbite). On constate une perte progressive de la couverture nominale: celle-ci n'est pas totalement récupérée à l'issue de chaque manoeuvre et la situation ne fait qu'empirer avec le temps.

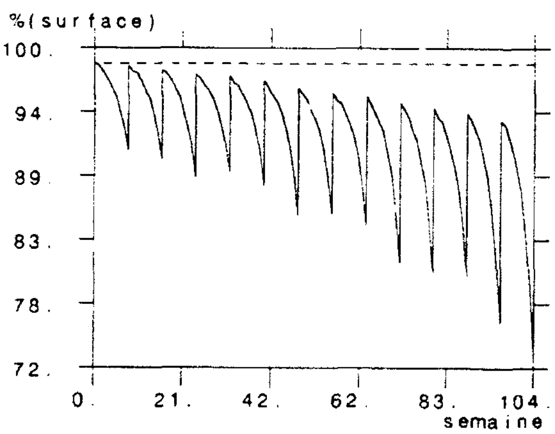

Figure 9: Evolution du pourcentage de couverture en fonction du temps. Corrections dans le plan de l'orbite.

Il s'avère donc nécessaire de réaliser également des corrections du plan de l'orbite (inclinaison et ascension droite du noeud ascendant). En ce qui concerne ce dernier paramètre, les manoeuvres réalisées se contentent d'assurer que les 3 plans d'orbite conservent la disposition relative qu'ils avaient initialement. En effet, sous l'influence des termes tesséraux du potentiel gravitationnel terrestre et sous celle de l'attraction luni-solaire, la vitesse de rotation des plans d'orbite diffère suivant les satellites. En revanche, la rotation d'ensemble des plans d'orbite, due essentiellement à l'aplatissement de la Terre et qui atteint environ 0.8° après 8 semaines, n'est bien entendu pas corrigée puisqu'il suffit de modifier légèrement le demi grand axe initial des orbites pour que la longitude des noeuds ascendants par rapport à la Terre reste fixe. Des manoeuvres de ce type effectuées toutes les 8 semaines permettent d'assurer une couverture qui reste toujours supérieure à 90% de la couverture initiale pendant les 7 années de durée de vie envisagées pour le système.

6.2.3 Coût du maintien à poste

Comme on vient de le voir, le maintien à poste envisagé nécessite de corriger la totalité des paramètres orbitaux. Néanmoins, la correction des noeuds ascendants ayant uniquement pour but de maintenir la disposition relative initiale des plans d'orbite, il suffit de manoeuvrer deux satellites sur les trois pour réaliser cette opération. L'identité des satellites à manoeuvrer peut être choisie de différentes façons. Pour cette première analyse, 2 scénarios de maintien à poste ont été envisagés:

a) on permute régulièrement le satellite qui ne manoeuvre pas.
b) on manoeuvre les satellites qui permettent de minimiser la somme totale de variation angulaire à réaliser.

Pour tester chacune de ces options, l'ensemble des logiciels mentionnés jusqu'à présent a été utilisé:

- intégration en éléments centrés des orbites de la constellation.
- calcul de la performance avant et après les manoeuvres de correction.
- calcul du coût minimal des manoeuvres de correction, modélisées par des transferts bi-impulsionnels.

Dans le calcul effectué, le coût des manoeuvres de phasage des satellites (affectant l'anomalie) a été négligé. Il est bien connu qu'il est possible d'obtenir le phasage désiré en réalisant une légère modification du demi grand axe de l'orbite, plaçant ainsi le satellite sur une orbite de dérive. Une seconde manoeuvre, faite en sens inverse de la première, permet de re-stabiliser le satellite une fois le déphasage souhaité obtenu. Le coût de cette manoeuvre dépend bien entendu de la durée tolérée pour l'obtention du phasage. Les premières évaluations réalisées ont montré qu'un délai de 24 heures permet de rendre le coût du phasage négligeable vis à vis du coût des autres manoeuvres, c'est pourquoi ce coût n'a pas été pris en compte par la suite.

Après avoir testé les deux scénarios de maintien à poste évoqués précédemment, il s'avère logiquement que le second est le plus économique. De plus, c'est celui qui assure la meilleure répartition des masses d'ergols consommées par les trois satellites.

Le coût total du maintien à poste a été évalué en réalisant une simulation complète sur 7 ans. Les résultats obtenus figurent dans le tableau de la figure 10.

On constate que le coût de maintien à poste s'avère relativement plus élevé que celui d'un satellite géostationnaire (85 m/s/an contre 50 pour un géostationnaire). Ce coût reste néanmoins tolérable. Il conviendra de vérifier s'il n'est pas possible, par un choix adéquat de la position des noeuds ascendants initiaux, de limiter le niveau des perturbations différentielles affectant les plans d'orbite et de réduire ainsi le coût global de maintien à poste.

Satellite	1	2	3
Coût annuel (m/s)	85	85	78
Coût total (m/s)	592	592	546
Masse totale d'ergols (% masse finale)	22	22	20

Figure 10: Coût du maintien à poste sur 7 ans

7. CONCLUSION

Le but de l'étude présentée dans cet article était de réaliser une analyse de la faisabilité d'un système de navigation fonctionnant suivant le principe du système GPS et assurant une couverture permanente de l'Europe et de l'Afrique à l'aide de 4 satellites seulement dont un géostationnaire.

Les paramètres orbitaux initiaux de cette constellation sont issus d'une étude antérieure[2] consacrée à la recherche de constellations à faible nombre de satellites. La constellation présentée ici présente plusieurs caractéristiques intéressantes:

- très faible nombre de satellites (4 satellites est le minimum requis pour assurer un service de navigation du type GPS),
- tous les satellites sont visibles en permanence de la même station de contrôle; il est ainsi possible utiliser les satellites comme de simples répéteurs d'un message de navigation élaboré au sol.
- orbites relativement "classiques".

L'étude de faisabilité présentée ici s'est surtout attachée à analyser les problèmes posés par la mise et le maintien à poste d'une telle constellation.

L'étude de la mise à poste a été faite en considérant diverses solutions:

- lancement individuel de chaque satellite, ou lancement couplé des 3 satellites non géostationnaires.
- injection préliminaire sur une orbite GTO standard (Geostationary Transfer Orbit) délivrée par un lanceur Ariane, ou injection sur une orbite dédiée à la mise à poste de ces satellites de navigation.

Il est apparu que l'orbite GTO standard Ariane constitue un très bon choix pour l'orbite d'injection, surtout si les 3 satellites non géostationnaires sont mis à poste à l'aide d'un lanceur unique. Dans ce cas, le coût de chaque transfert ne dépasse pas 2160 m/s en vitesse caractéristique. Il est cependant beaucoup plus élevé que celui d'un satellite géostationnaire (1500 m/s). L'orbite d'injection qui permet de minimiser la consommation d'ergols des transferts de mise à poste finale diffère peu de la GTO; elle conduit à une économie modeste (8% en masse).

Dans le cas où les satellites sont injectés sur GTO individuellement, le coût du transfert est très voisin de celui d'un satellite géostionnaire. Une telle procédure de mise à poste est cependant délicate car elle impose l'heure de tir des deux derniers satellites lancés.

L'étude du maintien à poste a nécessité dans un premier temps d'analyser l'évolution de la couverture du service de navigation sous l'influence des perturbations naturelles. Cette analyse a permis de montrer qu'un maintien à poste relativement fréquent s'impose (toutes le 8 semaines environ) pour assurer la permanence d'une couverture convenable.

Le coût des manoeuvres de maintien à poste nécessaire a ensuite été évalué sur la base d'une durée de vie des satellites de 7 ans. Cette première analyse semble indiquer que le coût annuel des manoeuvres s'élève à environ 85 m/s (contre 50 m/s pour un satellite géostationnaire).

Pour parachever l'analyse de mission d'un tel système, il conviendra désormais de vérifier s'il est possible de localiser de tels satellites avec une précision suffisante pour fournir une précision de navigation acceptable.

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Discussion

Question: Please provide the definition of TACSAT as intended in the paper presentation.

Reply: TACSAT is understood to be a theater satellite, that is to say, a system available where you want and when you want. This leads to consequences not only on the space platform but also on the ground support equipment.

Tactical Satellites for Air Command and Control

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This paper is divided in three parts corresponding to the following items:

- air command and control functions and defficiencies,
- contribution of TACSATS to air command and control,
- operational improvements due to TACSATS.

1. AIR COMMAND AND CONTROL FUNCTIONS AND DEFFICIENCIES

Air command and control functions are divided in four themes:

- surveillance,
- resources management,
- air activity control,
- intelligence.

1.1 Surveillance

Surveillance includes the generation and dissemination of the Recognized Air Picture (RAP). Three key issues are linked to this function, the detection of low observables and low flying objects, the ballistic missiles problem and the adverse satellites tracking.

a. Detection of low observable, low flying object

The current solution to detect low flyers is to use the contribution of radars on aircraft (AWACS). This type of platform may maintain its position during about eight hours, and you need four to five of them to maintain a round the clock surveillance capability.

The detection of low observables objects is solved by developing a sensing net, using different frequency range capability (visual, acoustic, infrared or electromagnetic) and different angle of presentation versus the possible target. This leads to a costly multiplication of sensors that must be in position on every possible penetration corridor.

b. Ballistic missiles

This new threat has to be included in the RAP. In order to do so, we have to detect the missiles in flight as soon as possible and track them during their flight.

A ground system like the PATRIOT has a limited detection capability coherent with the interceptor guidance.

A current satellite like DSP (Defence Support Program) was designed to detect a long range ballistic missile coming from a well known area. It has limited capability against a short range ballistic missile coming from a country involved with missile proliferation.

c. Adverse satellites

A theater commander has to know the location and coverage of the adverse observation satellites.

Today this function is performed by the Air Force and the satellites tracks are not included in the RAP, because this information is considered as a basis for threat evaluation or intelligence.

There is a requirement for the European countries to be able to fulfill this function with independent means.

1.2 Resources management

Resources management is divided in three themes:

- force management,
- airspace management,
- C2 resources management.

a. Force management

Force management includes the knowledge of our own forces state, the planning and tasking of air operations.

The key issues about this function are the interfaces with a multitude of other systems, the Air Task Order (ATO) preparation and dissemination and the weather forecasting.

In order to plan the air operations, a theater commander has to gather a multitude of informations coming from different systems like:

- the ennemy Order Of Battle (ODB) generated by intelligence system,

- his own forces state generated by Army, Navy and Air Force systems...

This leads to a requirement of connection with a multitude of other systems and the correct interpretation of the data collected by these systems.

Once this process has been performed, the Theater commander has to elaborate and disseminate the ATO to all the relevant units. These units are scattered on the Theater and may be connected by National means or Allied systems.

The planning function supposes for all the military operations that you are able to forecast the ennemy operations, your own operations and the weather. For this purpose you need adequate informations on the weather on the Theater and also outside the Theater.

b. Airspace management

All the air operations use a common mean that must be shared, it is the airspace. In order to do so you must be able to collect all the requirements in airspace coming from the different units (Air Space Request) and disseminate the Airspace Coordination Order (ACO) to all the relevant units.

This airspace planning is coordinated with the ATO planning and distributed three to four times a day to the units, this represent a hudge amount of messages spread all over the Theater.

c. C2 resources management

The air operations are performed using operators, ground-air-ground communication nets, consoles...
All these C2 resources must be allocated according to the current ATO corresponding to the Air Directives. This supposes a hudge information exchange between the different C2 entities. The process of these informations, and the planning of the best C2 resources allocation according to the Air Directives.

1.3 Air activity control

Air activity control is divided in two functions:

- air traffic control,

- air mission control.

a. Air traffic control

Air traffic control supposes first a flight regulation, that is to say an allocation of the corresponding airspace and C2 resources to the traffic requests. This supposes the collection of all the flight requests, the process of allocation in coordination with the airspace management function and the dissemination of the flight autorisations to the relevant units.

In addition, air traffic control is in charge of Search And Rescue (SAR) operations, mainly based on the accurate location of the friendly crews.

b. Air mission control

Once an air operation has been planned according to the ATO, this operation has to be controlled by air mission controllers, this supposes a good coordination between force management and air mission control function, that is to say a hudge amount of information exchange.

There is also an issue linked with radioelectric propagation phenomena, it is control of low flying aircrafts. This supposes a hudge amount of antennae wide spread along the theater to offer the communication means to air mission controllers.

Once a mission has been performed, the mission report must be prepared at the unit level and disseminated to the relevant operation centers for intelligence gathering and force management planning function.

1.4 Intelligence

Intelligence function is in charge of ennemy ODB assessment and forecasting and damage assessment. This supposes the gathering of all types of information: visual, electromagnetic ..., the fusion of these different informations and the dissemination of the relevant information to operating centers or units spread all along the Theater.

2. CONTRIBUTION OF TACSATS TO AIR COMMAND AND CONTROL

2.1 Surveillance

For surveillance function, TACSATS may provide queuing to the current sensor net by different means.
First ELINT or COMINT satellites may provide informations on ennemy airbases activity and trigger the AWACS take-off.
If these informations are not available, radars or electro-optics satellites may provide ennemy raids early detection in order to alert the Theater sensor net and activate the relevant sensors.

For the special case of ballistic missiles, launchers detection may be provided through radars or electro-optics satellites. Missile launch preparation may be detected through the use of ELINT or COMINT satellites picking-up the corresponding electromagnetic signals. In last ressort, missile launch detection may be detected by infrared satellites picking-up the plumes of the incoming missiles.

2.2 Resources management

For resource management, meteorological satellites may provide the relevant information in order to process the 24 to 48 hours forecasting necessary for the planning of air operations.
Broadcasting TACSATS are a good mean to disseminate the ATO message (force management) and the ACO (airspace management) to all the units widespread on the Theater.

Ground-to-ground TACSATS are a good mean to solve the operational requirements of:

- other systems database interrogation (force management),

- Airspace Control Means requirements diffusion (airspace management),

- C2 assets availability (C2 resources management).

The technical involved consideration are the quantity of information exchange and the widespread on the Theater of the different units involved in these functions.

2.3 Air activity control

Navigation TACSATS (signal location type) is the most efficient solution for aircraft localization, which is the major defficiency of search and rescue operations.

Ground-air-ground communication TACSATS is a good mean to solve the operational requirement of close control of low flyers, that is mandatory to allow an active control of low flying aircraft included in air mission control function.
Ground-to-ground communication TACSATS may solve the operational requirements of:

- flight plans dissemination (air traffic control),
- immediate airspace control means availability (air traffic control),
- rapid reallocation of planned missions (air mission control),
- rapid attack assessment (air mission control).

2.4 Intelligence

Ground-to-ground communication TACSATS are a good mean to solve the operational requirement of timely diffusion of filtered information to the relevant unit or operation centers in the Theater.
ELINT and COMINT TACSATS are the primary sensors to provide the ennemy ODB for SAM, radars, airbases, FLOT location and activity.
Radars or electro-optics TACSATS are a complementary platforms to solve the operational requirements of:

- ennemy ODB assessment,
- bomb damage assessment,
- air campaign results evaluation.

3. OPERATIONAL IMPROVEMENTS DUE TO TACSATS

3.1 Surveillance

TACSATS give the opportunity to implement a global extended air defence capability based on:

- active defence,
- passive defence,
- counterfire.

Active defence is alerted through the alert function derived from surveillance and threat evaluation, queued by the inflight missile track and controlled through the interception area prediction. All these functions may be performed by dedicated TACSATS which performances are refined according to the Theater ballistic threat.
Passive defence is alerted through the surveillance function and controlled through the impact area prediction. This control is mandatory in order to implement passive measures on limited area locations and during a limited time related with the impact time prediction.
Counterfire is alerted through the surveillance function and the weapon system queuing is performed through the launch zone determination. The adequate launch zone determination supposes dedicated TACSATS which performances are finely tuned according to the Theater ballistic threat characteristics.

3.2 Force management

Force management process is a complex and lenghtly process going from "general objectives" to "Air Task" and "Flight documentation" production.
The process that leads from general objectives to the ATO is performed in an operation center using modern means like artificial intelligence and knowledged based systems. Once the ATO has been elaborated it must be disseminated to the different units in the Theater. This requires a hudge exchange of information between operations centers and units. Communication TACSATS is a good mean to perform this information exchange.
With the ATO, the pilots in the units are able to perform the target attack preparation while the operation officers in the operation centers may perform the global mission preparation (deconfliction, flight routes ...).
Once the global mission preparation has been performed the pilots in the units may process the detailed mission preparation and issue the flight documents.
They are ready to take-off as soon as they receive the air task from the operation center.
This short comment about force management shows that by using adequate communication means and adequate intelligent process, we may reduce the global time required by the function. Communication TACSATS may provide adequate means to reduce the timelines involved in this process and drive changes in operational procedures.

3.3 Airspace management

The ACO is a complex message that defines the different areas in the airspace. This message includes air routes, transit corridors, low level transit routes, weapon free zones, air base defence zones, restricted operations zones, high density airspace control zone ...

It is a lenghtly process to elaborate this message and to disseminate it to the relevant units. Today this message is distributed every six to eight hours and valid for the same period. The consequence is that the Army by example has a dedicated airspace available for a six hours period to perform its operations. This has been a problem in the Gulf War where the French Army had a radar on a helicopter (HORIZON) that could not be used with its best performance because of the limited flight altitude.
TACSATS could provide technical means that may introduce new operational procedures like dynamic allocation of the airspace.

4. CONCLUSION

This short analysis has shown that communication TACSATS are needed to perform all the air mission command and control functions.
Different types of communication TACSATS are required:

- broadcasting,
- bidirectional,
- ground-to-ground,
- ground-to-air.

The development of TACSATS may change the current military procedures (especially the timelines for ATO or ACO).

TACTICAL SATELLITES FOR AIR COMMAND AND CONTROL

AIR COMMAND AND CONTROL SYSTEM FUNCTIONS (1)

AIR COMMAND AND CONTROL SYSTEM FUNCTIONS (2)

TACSATS FOR SURVEILLANCE

TACSATS FOR RESSOURCES MANAGEMENT

TACSATS FOR AIR ACTIVITY CONTROL

TACSATS FOR INTELLIGENCE

OPERATIONAL IMPROVEMENTS FOR SURVEILLANCE

OPERATIONAL IMPROVEMENTS FOR FORCE MANAGEMENT

OPERATIONAL IMPROVEMENTS FOR AIRSPACE MANAGEMENT

CONCLUSION

VU-GRAPH0

AIR COMMAND AND CONTROL SYSTEM FUNCTIONS (1)

FUNCTIONS	KEY ISSUES
AIRSPACE SURVEILLANCE	LOW OBSERVABLES, LOW FLYING OBJECTS BALLISTIC MISSILES ADVERSE SATELLITES
FORCE MANAGEMENT	INTERFACE WITH THE OTHER SYSTEMS (INTELLIGENCE, ELECTRONIC WARFARE ...) AIR TASK ORDER PREPARATION AND DISSEMINATION WEATHER FORECASTING
AIRSPACE MANAGEMENT	"REAL TIME" ALLOCATION OF THE AIRSPACE "AVAILABLE" AIRSPACE DISSEMINATION
C2 RESSOURCES MANAGEMENT	INFORMATION EXCHANGE WITH C2 ENTITIES

VU-GRAPH1

AIR COMMAND AND CONTROL SYSTEM FUNCTIONS (2)

FUNCTIONS	KEY ISSUES
AIR TRAFFIC CONTROL	FLIGHTS REGULATION COORDINATION WITH AIRSPACE MANAGEMENT SEARCH AND RESCUE OPERATIONS
AIR MISSION CONTROL	INTERFACE WITH FORCE MANAGEMENT ACTIVE CONTROL OF LOW FLYING AIRCRAFTS MISSION REPORT PREPARATION AND DISSEMINATION
INTELLIGENGE	ENNEMY ORDER OF BATTLE ASSESSMENT DAMAGE ASSESSMENT DISSEMINATION OF THE RELEVANT INFORMATION

VU-GRAPH2

TACSATS FOR SURVEILLANCE

DEFFICIENCIES	OPERATIONAL REQUIREMENTS	TACSATS TYPE
LOW OBSERVABLES, LOW FLYING OBJECTS	AIRBASES ACTIVATION RAIDS EARLY DETECTION	ELINT AND COMINT RADARS OR ELECTRO-OPTICS
BALLISTIC MISSILES	LAUNCHERS DETECTION MISSILE LAUNCH PREPARATION MISSILE LAUNCH DETECTION	RADARS OR ELECTRO-OPTICS ELINT AND COMINT INFRARED DETECTION
ADVERSE SATELLITES	COMMUNICATION SATELLITES POSITION OBSERVATION SATELLITES COVERAGE	

VU-GRAPH3

TACSATS FOR RESOURCES MANAGEMENT

DEFFICIENCIES	OPERATIONAL REQUIREMENTS	TACSATS TYPE
INTERFACE WITH THE OTHER SYSTEMS	OTHER SYSTEMS DATABASE INTERROGATION	GROUND TO GROUND COMMUNICATION
AIR TASK ORDER PREPARATION AND DISSEMINATION	ATO MESSAGES DISSEMINATION	BROADCASTING
WEATHER FORECASTING	24 TO 48 HOURS FORECASTING	METEOROLOGICAL
"REAL TIME" ALLOCATION OF THE AIRSPACE	AIRSPACE CONTROL MEANS REQUIREMENTS	GROUND TO GROUND COMMUNICATION
"AVAILABLE" AIRSPACE DISSEMINATION	AIRSPACE CONTROL ORDER DISSEMINATION	BROADCASTING
INFORMATION EXCHANGE WITH C2 ENTITIES	C2 ASSETS AVAILABILITY	GROUND TO GROUND COMMUNICATION

VU-GRAPH4

TACSATS FOR AIR ACTIVITY CONTROL

DEFFICIENCIES	OPERATIONAL REQUIREMENTS	TACSATS TYPE
FLIGHTS REGULATION	FLIGHT PLANS DISSEMINATION	GROUND TO GROUND COMMUNICATION
COORDINATION WITH AIRSPACE MANAGEMENT	IMMEDIATE AIRSPACE CONTROL MEANS AVAILABILITY	GROUND TO GROUND COMMUNICATION
SEARCH AND RESCUE OPERATIONS	AIRCRAFT LOCALIZATION	NAVIGATION
INTERFACE WITH FORCE MANAGEMENT	RAPID REALOCATION OF PLANNED MISSIONS	GROUND TO GROUND COMMUNICATION
ACTIVE CONTROL OF LOW FLYING AIRCRAFTS	CLOSE CONTROL OF LOW FLYERS	GROUND/AIR/GROUND COMMUNICATION
MISSION REPORT PREPARATION AND DISSEMINATION	RAPID AIR ATTACK ASSESSMENT	GROUND TO GROUND COMMUNICATION

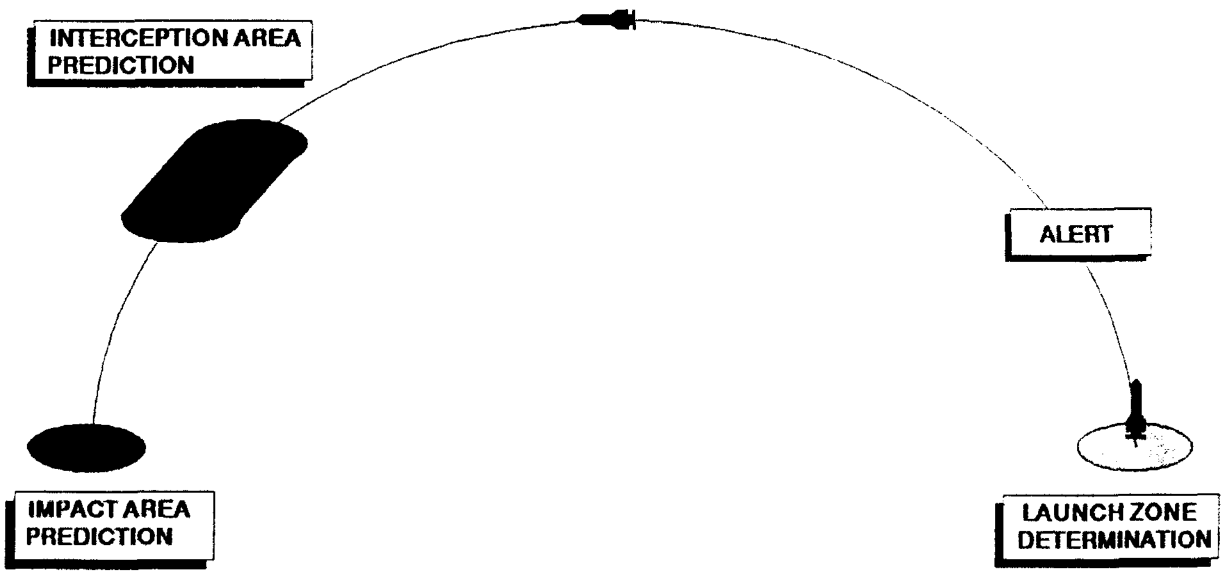
VU-GRAPH5

TACSATS FOR INTELLIGENCE

DEFFICIENCIES	OPERATIONAL REQUIREMENTS	TACSATS TYPE
ENNEMY ORDER OF BATTLE ASSESSMENT	SAM, RADARS, AIRBASES, FLOT LOCATION AND ACTIVITY	ELINT AND COMINT RADARS OR ELECTRO-OPTICS
DAMAGE ASSESSMENT	BOMB DAMAGE ASSESSMENT AIR CAMPAIGN RESULTS EVALUATION	RADARS OR ELECTRO-OPTICS
DISSEMINATION OF THE RELEVANT INFORMATION	TIMELY DIFFUSION OF FILTERED INFORMATION	GROUND TO GROUND COMMUNICATION

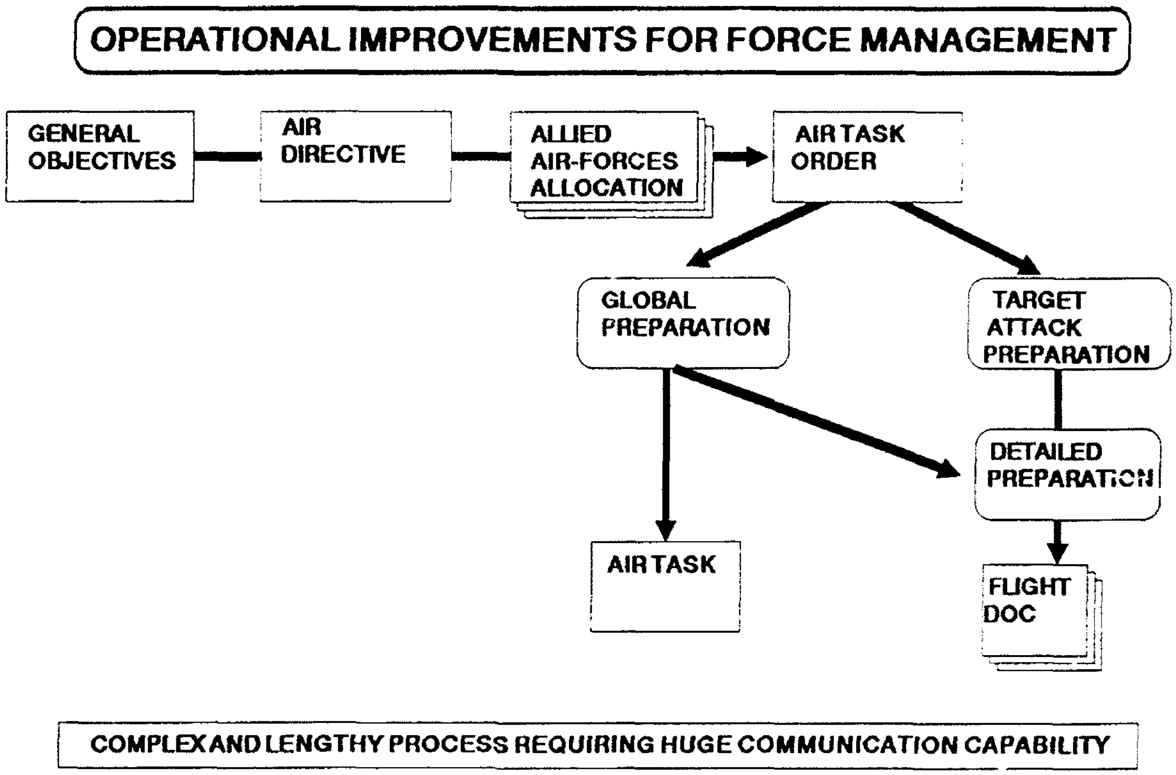
VU-GRAPH6

OPERATIONAL IMPROVEMENTS FOR SURVEILLANCE

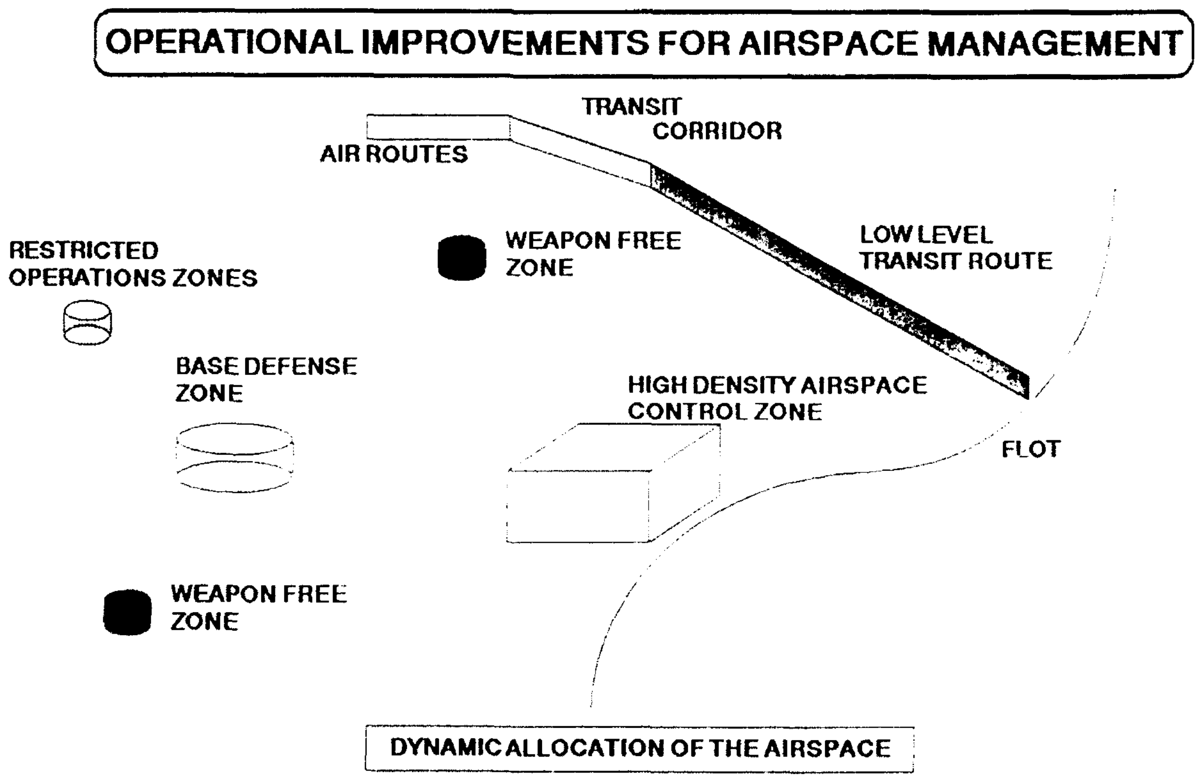


REAL EXTENDED AIR DEFENCE CAPABILITY

VU-GRAPH7



VU-GRAPH8



VU-GRAPH9

CONCLUSION

- COMMUNICATION TACSATS ARE NEEDED FOR ALL THE AIR COMMAND AND CONTROL FUNCTIONS

- DIFFERENT TYPES OF COMMUNICATION ARE REQUIRED : BIDIRECTIONAL
  BROAD CASTING
  GROUND TO GROUND
  GROUND TO AIR

- DEVELOPPEMENT OF TACSATS MAY CHANGE THE CURRENT MILITARY PROCEDURES

VU-GRAPH10

CONSIDERATIONS FOR NATO SATELLITE COMMUNICATIONS IN THE POST-2000 ERA

by

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1.The National Delegates Board of AGARD, upon recommendation by the Avionics Panel of AGARD, approved in March 1986 the establishment of WG-13 to study satellite communications for NATO under the direction of Prof Dr. Nejat Ince of Turkey.

2.Some 14 scientists/engineers, from research and industrial establishments of Canada, France, The Federal Republic of Germany, Norway, Turkey, the United Kingdom, the United States of America as well as from International Military Staff of NATO and SHAPE Technical Centre, participated in the work of WG-13.

3.This paper is a brief summary of the studies carried out by the group in the period 1988-1990 on the type of satellite communication systems which NATO can have in the post-2000 era including the critical techniques and technologies that need to be developed for this purpose.

4.In accordance with the Terms of Reference, the Group considered a time period beyond NATO IV and other national systems now in the implementation or planning stage, which would cover a time span of 20-30 years, i.e. 2000-2030. It was recognized that the earlier part of this period would be constrained by the existing and planned assets but the later part would be, and should be, more technology-driven. The following assumptions are made which take into account perceived trends and desirable attributes for future SATCOM systems:

i) The area of interest for NATO will remain as is to-day and will include the polar region

ii) The use of SATCOM will be more pervasive particularly for small mobile users (aircraft, land-mobile, ships and submerged submarines) to support general purpose and modern C3I structures.

iii) SATCOM will be integrated with the future ISDN networks now being planned and implemented in the nations and NATO. This may require SATCOM to have improved effective performance with respect to such parameters as delay and echo.

iv) The need for increased survivability against both physical and jamming threat will continue.

v) The use of frequencies in the EHF and optical bands for greater capability (e.g. AJ capability and communications with submerged submarines) and smaller terminals are foreseen.

vi) The future SATCOM systems will be required to be cheaper and more affordable.

vii) There will be the usual need for interoperability.

5.The Group agreed that the above attributes could be taken as inputs and goals for the system architectures to be developed for a future NATO SATCOM. In fact, these attributes were derived from the deficiencies of the present system which is not flexible enough with respect to growth in capacity and capability, and has a high degree of electronic and physical vulnerability and does not provide communications for the polar region and submerged submarines. The system development in the past has been based on successive discrete steps in capability and spending and each procurement has contained an important cost element of R&D. There has been a minimum of joint national R&D and use of the NATO system which resulted, among other things, in considerable interoperability problems.

It was agreed that what was required for the coming decades which may be characterized by "uncertainty" and "shrinking military budgets" was a very flexible, modular SATCOM system whose communication capacity and resilience to ECM and physical threat can be modified when operational requirements change, however, without having to undertake excessive R&D and total replacement of the space segment.

6.For the development of system architectures to achieve flexible and highly cost-effective SATCOM options for NATO, a technical survey has been made and information collected on the satellite system concepts being considered nationally (Canada, France, FRG, UK and USA) and internationally (ESA, Intelsat, Eutelsat, INMARSAT) for both civil and military applications as well as on related technological R&D activities and operational aspects regarding threat and environmental factors such as propagation and the usage of frequency spectrum.

7.The status of the following techniques/technologies and concepts which appear feasible and exploitable by future SATCOM systems and which are consequently being investigated nationally and internationally have been described in the report:

i) multi-beam/phased-array antennas with adaptive spatial nulling and multiple transmit spot beams,

ii) ECCM techniques,

iii) flexible and programmable on-board signal processing and switching techniques and devices,

iv) multi-frequency payloads,

v) multi-satellite systems to create spatial uncertainty for the enemy,

vi) use of tethers in space,

vii) laser and millimetre-wave communications for inter-satellite links,

viii) blue-green lasercom for submerged submarines,

ix) application of superconductivity, artificial intelligence, neural networks, robotics and of space-borne computers/software for signal processing and for manual and autonomous control of spatial and terrestrial resources,

x) power generation in space,

xi) spacecraft propulsion systems,

xii) launch vehicles and space transportations,

xiii) nuclear effects and hardening techniques.

xiv) physical attack and protective measures against directed energy beams (laser, particle, RF) and ASAT etc.,

xv) sensitive, light, long-life materials, components and devices for sensing, power generation, amplification and control.

8.The speed of progress made in the above areas will be determined mainly by the urgency of the need for, and the amount of resources allocated to, them. These technologies and new production methods coupled with basically software-controlled processing transponders with a capability to continuously adapt to changing requirements are expected to lead to more flexible and reliable, lighter and less-power consuming and altogether more cost-effective satellites than the present ones. Moreover these satellites can be launched by a number of different launch vehicles. Further reduction in cost may be obtained by sharing the satellites (single an/or cluster) between NATO and the Member Countries.

9.It can be stated generally and with confidence that in the time period in question it will be possible to design and build any satillite to meet almost any requirement. Technology exists or will be available for whatever communications performance and level of hardening is required as well as launch vehicles with capability to place the resulting satellite of whatever weight and power into any required orbit. The constraints will be the availability of orbital slots, frequency spectrum and, of course, funds.

10.The cost considerations have therefore been the driving factor for the systems reported here. When assessing different concepts for satellite designs and system architecture what is importanat is not so much their absolute but rather their relative costs. Accordingly a cost model of the satellite system has been established which takes into account:

- frequency band used (SHF, EHF),
- number of transponders,
- spacecraft reliability,
- R&D cost,
- power required,
- weight,
- launch cost,
- recurring cost,
- system availability.

11.Several SATCOM system architectures with the potential of meeting possible future NATO requirements implied in the paragraphs above have been defined using different orbits (geostationary, polar, 12-24-hr inclined at 63°.4 and Low Earth Orbit LEO) and a number of satellites with single and/or dual frequency transponders (SHF and EHF) which can be configured to meet any operational requirement. Table -1 lists the architectures considered in the report and gives the number of active spacecraft for full continuous coverage of the NATO area including the polar region as well as the total number of spacecraft needed for 7-years and 21-year periods for a certain given spacecraft reliability. Architectures based on the use of LEO and a combination of geostationary and polar-orbit satellites were eliminated from further consideration on cost grounds and the others were subjected to more detailed cost-performance analysis using the cost model mentioned in paragraph 10 above.

12.Table -2 lists some twenty different promising architectures (cases) for a future NATO SATCOM and gives the associated R&D, recurring and total costs for different spacecraft reliability and continuous service availability for 7 years.The common attributes of these architectures are the following(see Fig. -1) :

(a) i) The transponders have adaptive receive (with steerable nulls) and multi-beam transmit antennas (1 earth cover, 1 Europe cover, 1 polar spot and 2 steerable spots)

ii) A flexible channelization technique is used on board the satellites at both SHF and EHF. At EHF this is exploited in an on-board processing concept that prevents the satellite downlink transmitter from being loaded by the jammer and also to prevent unauthorized access to the satellite. For flexible AJ processing and ease of interoperability a full bandwidth (2 GHz at EHF, 500 MHz at SHF) filter band is provided using perhaps different filter technologies to obtain different selectivities required (see Fig. E-1) where the channelization can be controlled by telecommand to avoid interference, to alter the satellite capacity allocated to various geographical areas and to adapt the specific requirements due to restrictions in the tunability of the NATO or national ground segment. At EHF, where the flexible channelization technique is coupled with on-board processing for AJ purposes then switching between high-selectivity filter bank outputs (element filter output) will be performed at a high rate and controlled by an on-board transec equipment which can be programmable (in orbit) to support several simultuneous uplinks.

(b) The ground segment would consist of both SHF and EHF terminals. The SHF terminals would be those existing at the end of the NATO IV era and would be used mainly to support common-user trunks. General transition rom SHF to EHF is foreseen to take place over the period covered in the study to support mainly mobile/transportable users many of which may have demanding AJ and/or LPI requirement.

The EHF ground segment:

i) has preferably non-synchronized frequency-hopped (because of its better performance in disturbed and time-variant propagation conditions and better suitability to small terminals than the direct sequence modulation system) terminals operating in FDMA with flexible data rates and redefinable codes.

ii) consists of the simultaneous accesses (for system comparison purposes) given in Table -3.

(c) The systems

i) have the virtue of allowing easy transition from existing to future architectures,

ii) Have minimum development, recurring and launch costs,

iii) are upgradable and expandable on a scale to meet operational requirements,

iv) defective and life-expired elements of the system are replaceable without man intervention,

v) spacecraft are capable of being refuelled without man intervention,

vi) have virtually zero down-time at low cost,

vii) make maximum use of orbital slot allocations,

viii) allow spatial distribution of spacecraft to reduce their vulnerability to jamming and physical attack.

It should be noted that the data in Tables -2 (a) and (b) are for a 7-year period. During the 21-year total period, three stages of complete space segment replacement are expected to occur, which would allow for an update for changes in traffic or other requirements. For dual frequency systems development cost will be incurred at each stage.Single frequency systems will not incur such costs since the same designs of spacecraft would be used throughout the 21-year period; only the mix of EHF and SHF types would change. Provided military components are used in the design of the spacecraft it should be possible to maintain full availabilty over the 21-year interval.

13.An examination of the data shows that:

a) For geostationary operations the cost of interconnection of spacecraft is of the order of 3 % , and is not more than 4.5 % for the inclined orbits (Tundra as the most expensive case).Interconnection provides significant improvement in service availability probability, ranging from 0.14 at the lower inherent spacecraft probabilities to 0.04 at the high end.

b) Increasing the space segment availability by the amount given in (a) above without, however, using Inter-Satellite Links ISL would require launching more satellites and this would increase the system cost by about 25 %.

c) Operation in inclined orbits costs about 50 % more than the geostationary case for the same service availability, but gives full NATO coverage including the polar region.

d) The geostationary case 1 corresponds to the NATO IV satellite as far as coverage and the number of satellites and reliability are concerned. It is interesting to note, however, that the 7-year system cost of Case 1 and that of NATO IV (about 400$M) are almost identical even though Case 1 satellites have considerably more capacity (in SHF and EHF) and significantly greater resistance to jamming (on-board signal processing in EHF and adaptive nulling antennas).

e) The system cost changes significantly with the 7-year service availability probability. How many satelittes would be needed for a 21- year period without having excessive capacity would depend on this as well as on what residual capacity would remain at the end of each seven-year period and how the change in requirements is introduced; abruptly at each 7-year period or progressively during the 21-year period. In the latter case, some reduction in the total number of satellites required and hence in total cost would be expected.

14.The architectures which appear cost-effective and promising are given in Table -4.

The following comments can be made about these architectures:

a) An adequately wide range oef architectural options are presented from which the architecture best suited to the requirements, as they will be known nearer the date of system implementation, can be selected.

b) Based on the assumptions made regarding possible future NATO requirements, reliabilities of future electronic systems and costs per kilogram of Payloads, Spacecraft Platforms and Launches which have been used consistently for all of the candidate architectures, it can be concluded that:

i) Provided NATO can accept the coverage provided by a geostationary only system of satellites, Architecture A is the lowest cost solution.

ii) If polar coverage obtained by leasing from the USA, costs less than Cost (H-A) or Cost (I-A) then Architectures A or B plus Polar leasing would provide the next lowest cost options. Option B gives improved availability and AJ capability but at 33 % higher cost than the cost of A.

iii) The lowest cost architecture which provides full coverage is Architecture H at a cost increase of 50 % over geostationary only (Case B).

iv) For a further 5 % increase in cost, an improvement from 0.95 to 0.98 in operational availability and an enhanced AJ capability can be obtained by using Architecture I. This architecture is probably the most cost-effective option of those considered, to all of the assumed future NATO requirements.

15.It is likely that cost will be the driving factor in determining the hoice of a future SATCOM architecture and it is therefore appropriate to consider the three dominant cost factors (R&D, replacement and launch costs) and indicate what steps could be taken to bring about cost reduction in each case.

a) Economy in R&D costs could be obtained through NATO/National collaboration and by adopting a modular approach to system diversification and evolution. An effective way of achieving the latter would be to develop at the outset separate SHF and EHF spacecraft and use them, in GEO and TUNDRA orbits alike, in a mix determined by the changing requirements.

b) The use of smaller spacecraft, even though more of them may be needed, could lead to lower system costs because of the economies of scale. Such economies of scale would be further enhanced if the same spacecraft types are used at all stages of system evolution over a period of, say, twenty years. They will also be enhanced if the same spacecraft types are bought for national as well as NATO use.

c) Launch costs can be minimized by reducing spacecraft mass, in particular through the exploitation of new technology. It is also important to maximize compatibility with the largest possible range of launch vehicles.

d) Interconnection of spacecraft increases system reliability and therefore tends to reduce the total number of spacecraft that need to be launched.

e) Finally, long-term planning is the key to achieving reductions in both R&D and recurring costs.

16.The NATO SATCOM systems so far acquired have been based on national developments adapted to NATO requirements and the continuity of service (not necessarily full service) has been obtained by sharing or borrowing capacity from national systems. The national systems, in turn, have relied for continuity of service on the availability of capacity on the NATO system. Each procurement has contained an important element of R&D costs and since successive systems have been developed almost independently of each other, R&D costs have been, like the replacement cost, also recurring

There has been a minimum of joint national R&D and use of the system and each procurement has been preceded by lengthy negociations on production sharing which has not, in general, satisfied, at least, some of the member countries. As a result of having independent NATO and national systems there has been considerable interoperability problems.

It is believed that this trend, based on successive jumps in spending and capabilty with a minimum degree of general national participation should be and can be changed to meet the needs of the coming decades which may be characterized by uncertainly and shrinking military budgets requiring affordable and flexible systems.

The member countries have adequate experience within NATO and Europe and know that under these circumstances it is necessary to resort to joint R&D, procurement and use of the system while ensuring effectiveness and competitiveness for keeping the costs down.

17.What needs to be done jointly are:

a) To define NATO and national requirements for satellite communications.

b) To develope and agree on a system architecture.

c) To delineate those technological areas which are critical and require R&D.

d) To encourage and support companies and R&D establishments to form research partnership for development and production to be carried out in a competitive manner.

It is believed that the architectures evaluated and recommended in this report form a good foundation for (b) above and ensure also that the satellite designs outlined that are flexible need not change basically over a period of some twenty years or longer thus keeping the R&D and recurring costs to a minimum.

The report outlines also certain critical technologies for (c) above which need R&D. Some of these R&D topics are common to military and civilian satcoms; some others which are specific to military, are likely to be common to both national and NATO systems and yet another category of topics will be NATO-specific. It would therefore be necesarry to make a more detailed assessment of the R&D topics and determine where R&D is a prerequisite and either can be relied upon present/future civilian developments or carried out jointly by member nations.

18.The following areas appear as first candidates for a NATO R&D effort because they would provide solutions to problems which are NATO-specific and can be made available within the time-frame considered for NATO SATCOM systems.

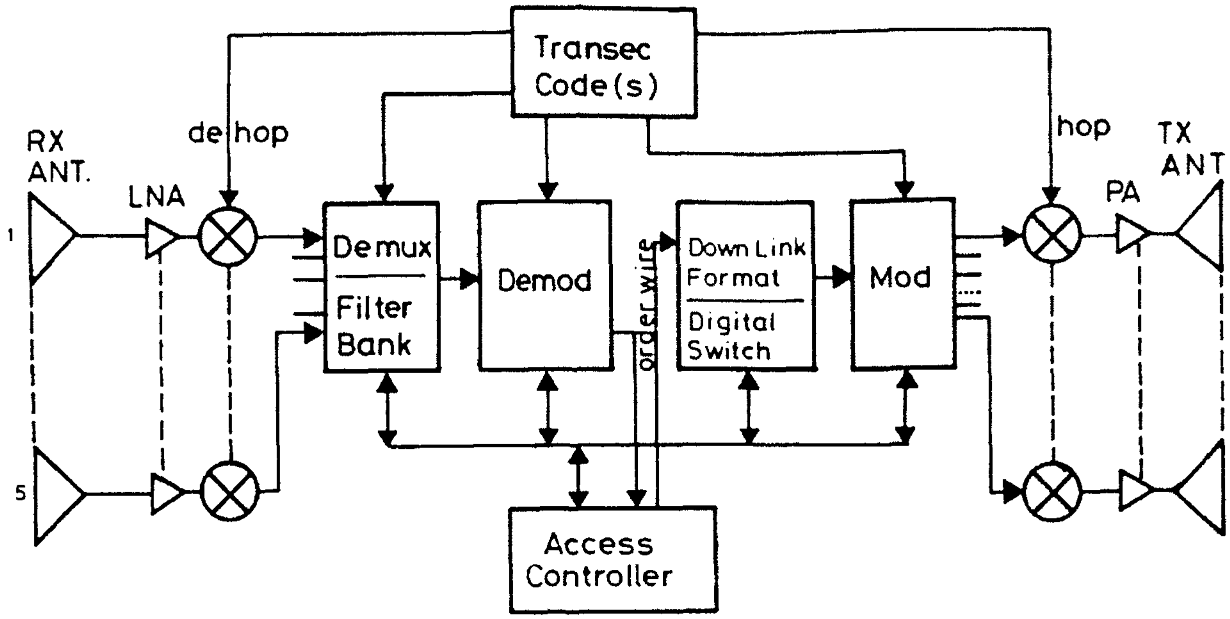
a) On-board flexible anti-jam signal processing which can be controlled by software to meet AJ requirements generally and to adapt to national modems and new modems introduced during the lifetime of the satellite(s).

b) Adaptive nulling AJ receive antennas tailored to NATO needs and to keep costs down.

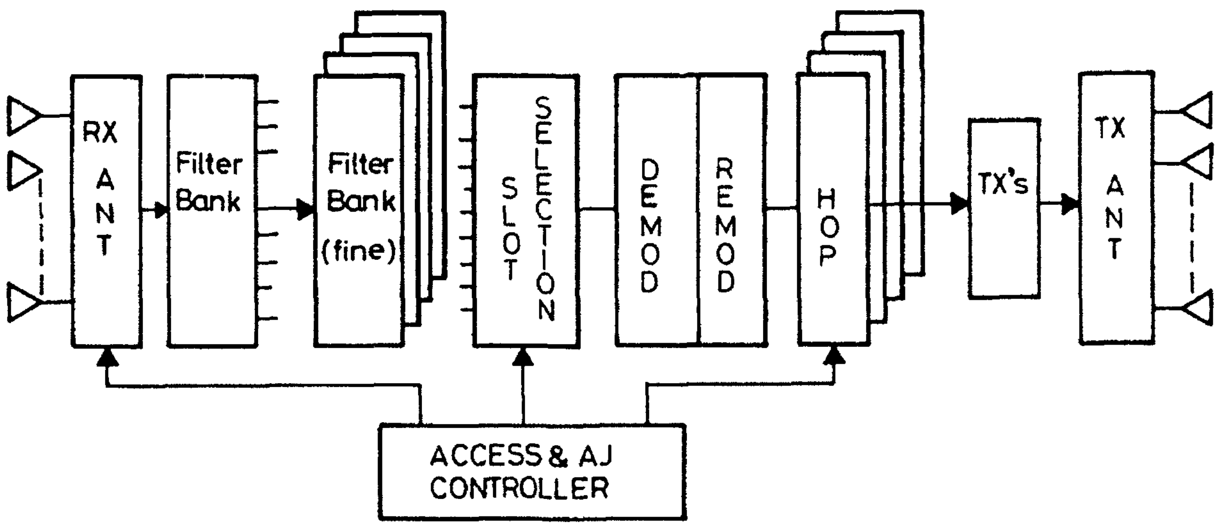
c) Autonomous control of the spacecraft and O&M generally using techniques of artificial intelligence, neural networks and robotics.

By sporing R&D activities, limited to payload technology, in the member nations even on modest scale, NATO can expect to get a better insight into and to make an impact on current technology developments carried out for civilian and military purposes.

19.It is believed that the collaborative approach outlined above for SATCOM acquisition in NATO give tasks to all the existing bodies in NATO such as NACISA, STC, DRG, IEPG, AGARD, etc., and would probably not necessitate creating new structures. Cost and benefit anlysis carried out in the report show that the architectures recommended can be implemented in the manner suggested above and could lead to systems which are considerably cheaper and much more effective than those we have had so far.



a) Non-transparent satellite with about 50 MHz information bandwidth hopped across 2000 MHz. All terminals hop in synchronism. There will be as many dehoppers in the satellite as there are different nets with different codes.



b) Transparent satellite (full bandwidth) allowing operation with non-synchronised frequency-hopped terminals.

Fig. -1 Two promising AJ satellite designs with on-board processing and routing

TABLE -1
Number of Satellites
Required for different SATCOM Architectures

Features Architecture	No of Active S/C for full continuous Coverage	Comment	No of S/C for 7 years	Total No of S/C for 21 years
(a) Proliferated LEO	240		>240	>500
(c) Inclined Elliptical Orbit (c1) LOOPUS (c2) MOLNIYA (c3) TUNDRA	 9 3 x 12 - hrs 2 x 24 - hrs 2 x 24 - hrs		 27 9 6 6	 81 27 18 18
(d) GEO	1	Baseline	3	9
(e) Systems of Satellites in more than one Orbit (e1) GEO + Polar	 1 GEO + 6 Polar	 S/C in GEO are dual frequency single in Polar orbit	 3 GEO + 18 Polar	 9 GEO + 54 Polar
(e2) GEO+24-hr MOLNIYA (a) Dual freq S/C in GEO+single in Inclined Orbit (b) Single freq in both orbits	 1 GEO + 2 inclined 2 GEO + 2 inclined	 GEO S/C have EHF and SHF Inclined EHF or SHF GEO S/C are air of EHF and SHF Inclined EHF or SHF	 3 GEO + 6 inclined 6 GEO + 6 inclined	 9 GEO + 18 inclined 18 GEO + 18 inclined
(f) CLOUDSAT with 10dB ECCM advantage relative to (e2)b	Receive S/C 20 GEO. 20 Inc. TX S/C:- 2 GEO+2 Inc	Numbers will depend on jamming threat at the time	60 GEO 60 Inclined 6 GEO 6 inclined	180 GEO 180 inclined 18 GEO 18 inclined
(g) MEWS	The basic concept applies to all case (c) thro' (f)			

Table -2 (a) System Evaluation Data for Geostationary Operations (7 years)

Case	Spacecraft		Payload			Availability		Total No. Operating Spacecraft	System Costs			
	No / Orbit	Frequency	EHF	SHF	Connect	S/C	Space Segment		R & D	Recur	Launch	Total
1	2	Dual	Y	Y	No	0.61	0.85	2	242	82	76	400
2	2	Dual	Y	Y	Yes	0.61	0.93	2	262	87	79	428
3	3	Dual	Y	Y	No	0.61	0.94	3	242	163	115	520
4	3	Dual	Y	Y	Yes	0.61	0.98	3	262	173	118	553
5	4	Single	Y	Y	No	0.72 0.76	0.87	4	290	95	92	478
6	4	Single	Y	Y	Yes	0.72 0.76	0.93	4	321	106	96	523
7	6	Single	Y	Y	No	0.72 0.76	0.96	6	290	192	138	620
8	6	Single	Y	Y	Yes	0.72 0.76	0.99	[illegible]	321	212	144	[illegible]

Table - 2(b) Evaluation Data for 24-Hour TUNDRA Orbits (7 years)

Case	Spacecraft		Payload			Availability		Total No. Operating Spacecraft	System Costs			
	No / Orbit	Frequency	EHF	SHF	Connect	S/C	Space Segment		R & D	Recur	Launch	Total
1	2	Dual	Y	Y	No	0.61	0.72	4	223	221	126	570
2	2	Dual	Y	Y	Yes	0.61	0.86	4	238	236	130	604
3	3	Dual	Y	Y	No	0.61	0.88	6	223	368	189	780
4	3	Dual	Y	Y	Yes	0.61	0.96	6	238	393	195	826
5	4	Single	Y	Y	No	0.72 0.76	0.76	8	258	255	155	668
6	4	Single	Y	Y	Yes	0.72 0.76	0.87	8	288	285	163	736
7	6	Single	Y	Y	No	0.72 0.76	0.93	12	258	425	233	916
8	6	Single	Y	Y	Yes	0.72 0.76	0.97	12	288	475	245	1008
9	1	Single	Y	-	No	0.72	0.52	2	103	34	33	170
10	2	Single	Y	-	No	0.72	0.85	4	103	102	66	271
11	2	Single	Y	-	Yes	0.72	0.93	4	118	117	70	305
12	3	Single	Y	-	Yes	0.72	0.99	6	118	196	104	418

Table -3 : The EHF Ground Segment Assumed for System Comparison

Type of Terminal	Antenna Dia. (m)	Tx Powers (kW)	Transmission Rate (Baud)	Number of simultaneous accesses
Ship - Borne	1.0	0.1	2400 - 9600	20
Aircraft	0.5	0.1	2400	10
Submarine	0.25	0.1	100 - 2400	1
Land transportable	5.0	1.0	4 x 64000 (*)	15
	2.0	0.5	4 x 2400	30
Man - pack	0.5	0.01	75	30

(*) 16 kb/s codecs (adaptive sub-band coding) exist today with quality which equals that of the 64 kb/s PCM. It is expected that in the timeframe considered in this report there will be 8 kb/s or even lower - rate codecs available for use in SATCOM with qualities comparable to that of 64 kb/s PCM voice.

TABLE -4

COST-RELIABILITY COMPARISON OF CANDIDATE ARCHITECTURES FOR A SYSTEM LIFETIME OF 21 YEARS

Architecture	Cases No		Total No of operating spacecraft	No of Payload		Orbit / Freq.		Service Availability	Cost (21-year) ($M)			
	T	GEO		EHF	SHF	T	GEO		R & D	Recur.	Launch	Total
A	-	1	2	2	2	-	D	0.85	3 x 242 = 726	3 x 82 = 246	3 x 76 = 228	1200
B	-	4	3	3	3	-	D	0.98	3 x 262 = 786	3 x 173 = 519	3 x 118 = 354	1659
C	2	-	4		4	D	-	0.86	3 x 238 = 714	3 x 236 = 708	3 x 130 = 390	1812
D	4	-	6	6	6	D	-	0.96	3 x 238 = 714	3 x 393 = 1179	3 x 195 = 585	2478
E	6	-	8	4	4	E,S	-	0.87	288	11 x 95.01 = 1045	12 x 40.8 = 489.4	1822
F	8	-	12	6	6	E,S		0.97	288	1615	734	2637
G	12	4	9	9	3	E	D	0.97	118 + 786 = 904	665 + 519 = 1184	313 + 354 = 667	2755
H	12	7	12	9	3	E	E,S	0.95	290.5	1470.2	727.1	2488
I	12	8	12	9	3	E	E,S	0.98	320.86	1550.9	745.3	2617
J	8	8	18	9	9	E,S	E,S	0.96	320.86	2558.2	1166.1	4045

ADVANCED TECHNOLOGIES FOR LIGHTWEIGHT EHF TACTICAL COMMUNICATIONS SATELLITES*

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1. ABSTRACT

The communications capabilities provided by EHF satellites can range from low data rate services (75 to 2400 bps per channel) to medium data rate links (4.8 kbps to 1.544 Mbps per link) depending on the payload configuration. Through the use of EHF waveform standards, the EHF payloads will be compatible with existing and planned EHF terminals. Advanced technologies permit the development of highly capable, lightweight payloads which can be utilized in a variety of roles. The key payload technologies include adaptive uplink antennas; high speed, low power digital signal processing subsystems; lightweight frequency hopping synthesizers; and efficient solid-state transmitters. The focus in this paper is on the signal processing and frequency generation technologies and their application in a lightweight EHF payload for tactical applications.

2. INTRODUCTION

A motivating factor for the transition to EHF communications (i.e., 44 GHz uplinks and 20 GHz downlinks) is the requirement for improved interference protection with small mobile and transportable terminals for tactical and strategic users. Emerging technologies allow EHF communications systems, which can support both low data rate (LDR) and medium data rate (MDR) services, to be implemented in lightweight, low power configurations. Standard EHF payloads, utilize both advanced antenna systems and on-board signal generation and processing techniques to improve performance and protection. A payload which utilizes these features for a theater coverage application is described in this paper. The application of advanced signal generation and processing technologies to this lightweight payload result in a payload which can be incorporated into satellites of many sizes, ranging from large, multiple function satellites to small, augmentation satellites [1-4].

3. ANTENNA OPTIONS

Advanced antenna features for tactical applications can include the ability to provide a variable coverage uplink spot beam pattern, an autonomous nulling capability within the uplink spot beam pattern, or both. Variable beamwidth EHF antennas can be utilized in a variety of applications as shown in Fig. 1. For payloads in elliptical orbits, a variable beam width feature can be utilized to maintain an essentially fixed coverage area independent of satellite altitude [4]. In a geosynchronous orbit application, the beam variability can be used to satisfy different coverage and gain requirements such as in supporting tactical theaters of varying size and capacity requirements. One approach to obtaining the beam variability is by employing multiple feeds in the antenna. These feeds are combined with a variable power combiner network before going to the receiver. With 7 uplink feeds in the antenna, a 3-to-1 variation in beamwidth can be achieved, while 19 uplink feeds give a 5-to-1 variation. By incorporating both phase and amplitude control in the beamforming network and including a processor, the variable beamwidth antenna can also include autonomous nulling [5].

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4. LIGHTWEIGHT SIGNAL GENERATORS

In order to provide effective interoperability, it is important for all the EHF payloads to work with the same type of user terminals. Standard EHF transmission formats and dynamic access/configuration control are important features in providing this interoperability. The standard EHF waveform requires the dehopping and demodulation of communications signals on-board the satellite. As shown in Fig. 2, lightweight frequency hopping synthesizers can be implemented using direct digital synthesis techniques along with high-speed, hybridized bandwidth expansion circuitry. These advanced frequency generators yield almost an order of magnitude reduction in weight over frequency synthesizer subsystems of the early 1980's while also requiring significantly less than half the power.

The key design criteria for a payload frequency synthesizer are a low power configuration which has the ability to generate signals with low spurious frequency content while meeting the frequency switching speed requirement. These factors are key in selecting the bandwidth expansion approach as shown in Fig. 3. A switched filter bank approach is a straightforward implementation. However, filter size limits the number of frequencies (N) which can be selected for mixing with the direct digital synthesizer (DDS) output, thus impacting the amount of bandwidth which must be generated by the DDS. This adversely affects both the power and spur constraints by requiring a higher power DDS and by generating larger spurs. The alternative approach shown in Fig. 3 was selected for the payload frequency synthesizer. This approach utilizes a high speed phase-locked loop to generate the set of N frequencies which are mixed with the DDS output to expand the bandwidth. The wide bandwidth of the loop allows N to be large, thus requiring a smaller DDS bandwidth. The reduced DDS bandwidth allows the use of a low power, CMOS based DDS and results in lower spur levels. The key technology challenges in implementing the wide bandwidth loop are the high-speed counter and the custom voltage controlled oscillator (VCO).

Some of the test results from a breadboard synthesizer which utilizes the wide bandwidth loop for bandwidth expansion are shown in Fig. 4. The breadboard synthesizer meets the switching speed requirement by settling to within 7.5° of the final phase in a little over 0.8 µsec. The goal for spur levels is also met by the breadboard synthesizer. A typical output spectrum is shown in Fig. 4.

5. HIGH-SPEED SIGNAL PROCESSORS

High-speed digital signal processing advances can be utilized to provide lightweight, low-power demodulators and signal processing subsystems capable of supporting many LDR and MDR channels. In the early 1980's, the use of combined analog and digital processing technologies provided the most efficient implementation for demodulating the LDR uplink waveform. As shown in Fig. 5, the frequency demodulation was performed by a surface acoustic wave (SAW) demodulator followed by a digital communications and acquisition processor. Now however, the progression of digital technology has advanced the state-of-the-art to the point where an all digital approach is more efficient. The development of application specific ICs (ASICs) which process the standard EHF waveforms will contribute significantly to the reduction in weight and power required by these subsystems. For further reductions in weight and power, the ASIC devices can be integrated into multi-chip modules to achieve the benefits associated with a wafer-scale level of integration as shown in Fig. 5. In this comparison of an LDR signal processor using multi-chip modules with an LDR signal processor from the early 1980's, an order of magnitude reduction in weight is obtained while the power is decreased by more than half for the same number of channels processed.

The digital Fast Fourier Transform (FFT) demodulator is implemented with two ASIC designs shown in Fig. 6. The sampled signals are first preprocessed before the actual transform is performed. In this FFT preprocessor chip, the signals are windowed, coherently integrated for adjustment of frequency sample spacing, and stored in memory. The heart of the demodulator is the in-place FFT chip shown in Fig. 6. The data is stored in memory, the FFT butterfly operations are performed, and after the transform is completed, the I and Q samples are converted to magnitude values for further processing by the uplink processor. The FFT chip is designed for use with data or acquisition channels and can perform a transform of up to 256 points.

The ASIC for the in-place FFT, shown in Fig. 7, has been designed, fabricated, and tested. The FFT ASIC is designed to meet the primary requirements for a space application: radiation hardness, high performance, low power consumption, and high reliability. A significant design feature in this chip which enables high performance in a low power configuration is the use of on-chip memory (RAM and ROM) for the data being processed and for the coefficients used in the FFT. The amount of memory required is minimized by employing an innovative in-place algorithm using dual port RAM.

The FFT chip, the preprocessor chip, and a communications uplink processor (CUP) ASIC can be configured into a communications demodulator, which is capable of processing up to 16 channels, as shown in Fig. 8. These ASICs, along with the supporting chips (A/D converters, CUP RAM, and mission ROM) can be packaged into a multi-chip module which is about 2" x 3" in size.

For MDR channels, similar ASIC technology is expected to yield efficient implementations for these higher data rate channels as well. A preliminary design for a four channel MDR subsystem requires three individual ASIC designs. Four demodulator chips are utilized in conjunction with a clock generator chip and an MDR processor chip to form the four channel MDR subsystem. The designs for the clock generator chip and the MDR processor chip allow cascading to support additional MDR demodulators for payloads with more than four channels.

6. EXAMPLE EHF PAYLOAD FOR GEOSYNCHRONOUS ORBITS

An example EHF payload, which utilizes a pair of variable beamwidth spot beam antennas and both LDR and MDR signal processing, is shown in Fig. 9. For this example payload, the uplink spot beam antennas utilize 7 feeds each and the downlink beams are formed using 1 feed each. Both LDR and MDR channels are supported in the spot beams. In addition, the payload provides LDR earth coverage service through a pair of earth coverage horns. The LDR processor supports 16 communications channels in each of the beams using the EHF common transmission format. The MDR processor provides a total of 4 channels of service in the spot beams with any mix between the two beams.

The main considerations in selecting a spot beam size are the required gain and the coverage area provided by the beam. The 1° to 3° spot beam size in this example payload, along with the 6 W solid state transmitter, will support 2.4 kbps service to a small terminal (2'/2W) while in the wide beam mode and will support 1 Mbps links to a medium size terminal (4'/12W) while in the narrow beam mode. The payload in Fig. 9 is estimated to weigh about 200 lb and require about 290 W (these estimates include 20% margins).

The 6 W transmitter and the 20" spot beam antennas provide sufficient EIRP to support both LDR and MDR links in a variety of modes and data rates with the total throughput for the payload depending on the mix of LDR and MDR terminals in a scenario. An example loading scenario is shown in Fig. 10. For this example, three types of terminals were assumed: a 6', 25 W ground terminal; a 4', 12 W transportable terminal; and a 2', 2 W portable terminal. The ground terminal is supported by the earth coverage beam in the example. The portable and transportable terminals are supported in the spot beams. One of the spot beams is set to a 3° beamwidth (about 1200 mile diameter coverage at the subsatellite point) while the other spot beam is set to a 1° beamwidth (about 400 mile diameter coverage at the subsatellite point). In this example, 27 LDR networks and 17 MDR links are supported for a total payload throughput of 3277 kbps.

A range of payload capabilities can be implemented using the variable beamwidth antennas, nulling processors, and the other key technologies described briefly in Fig. 1 and 2. These technologies can be used to implement small EHF payloads as in the example presented here. However, the same technologies can also be used in secondary anti-jam payloads or multiple function anti-jam payloads on large satellites as shown in Fig. 11. In addition, many of the same signal processing and frequency generation technologies are applicable for improving EHF terminals as well.

7. SUMMARY

A key feature for the flexible use of the EHF payload is to provide the ability to configure the payload to provide a variety of services. Supporting either LDR, MDR, or both types of channels in a variable beamwidth antenna helps provide this sort of flexibility to meet a broad range of user requirements. Development of the critical technologies for use in these types of payloads has been initiated. The technology areas include variable beamwidth antennas, lightweight frequency synthesizers, and high speed signal processors for both LDR and MDR channels.

8. ACKNOWLEDGEMENTS

The concepts which are presented here were developed through the MILSATCOM concept and technology development efforts of the lightweight EHF payload program at MIT Lincoln Laboratory. We thank the individuals who have contributed to this work. We especially thank John Drover for his work in developing the FFT ASIC and David Materna for his work in leading the design of the frequency synthesizer.

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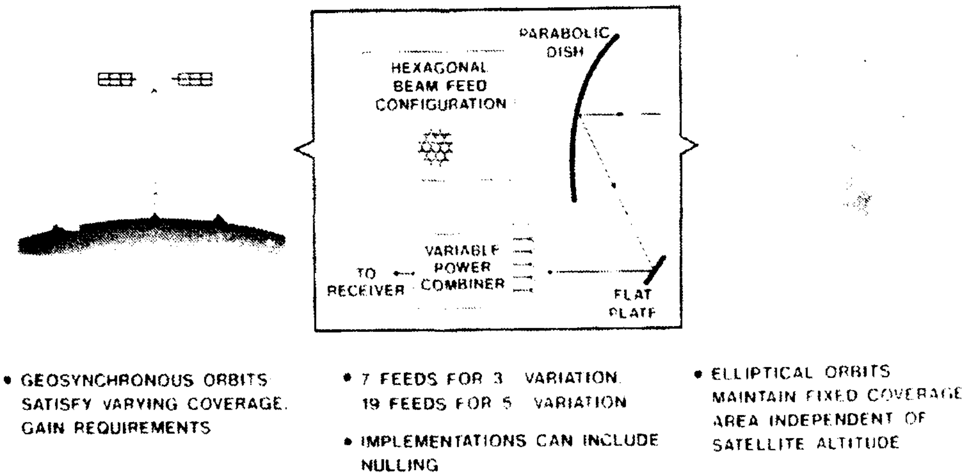


Figure 1. EHF Variable Beamwidth Antenna

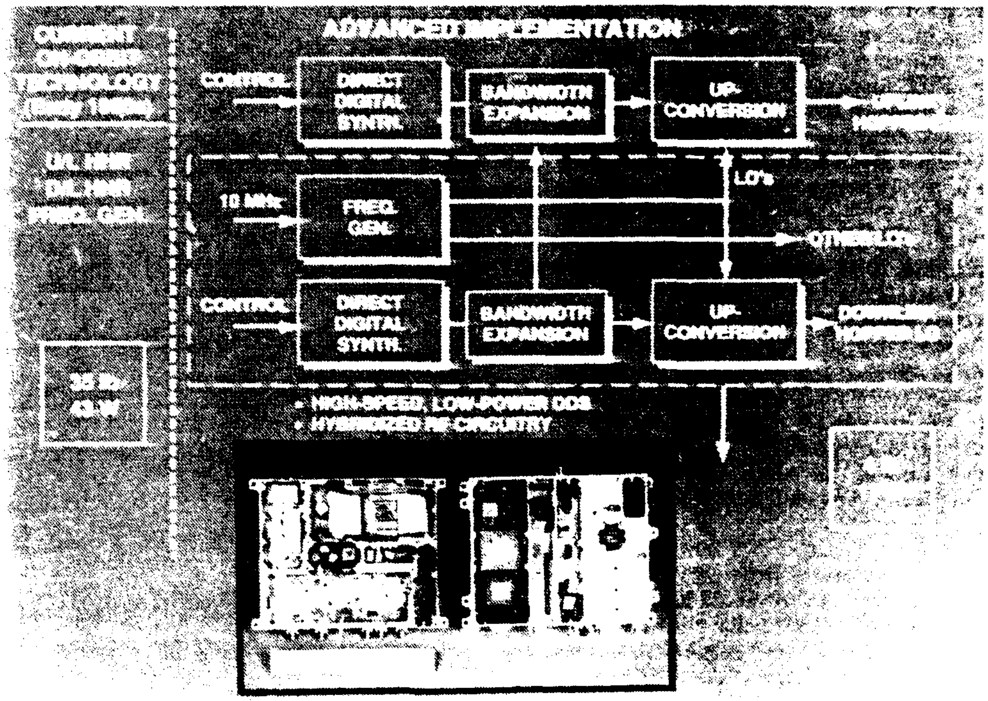
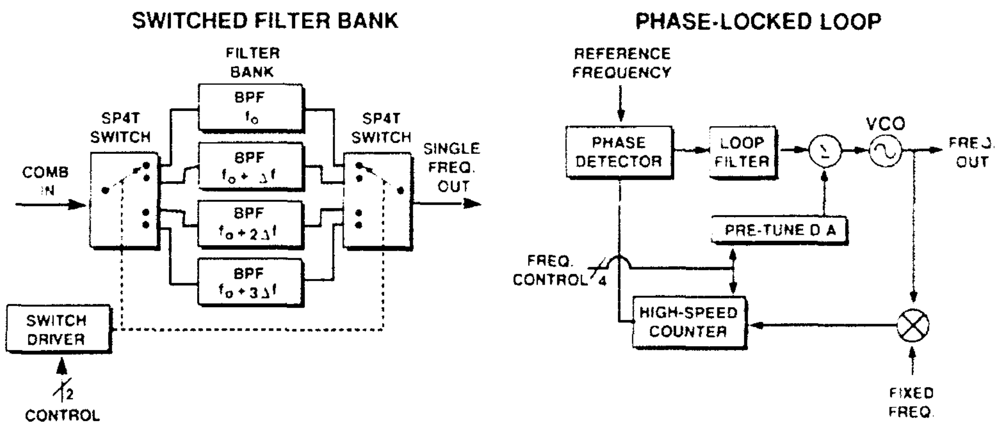


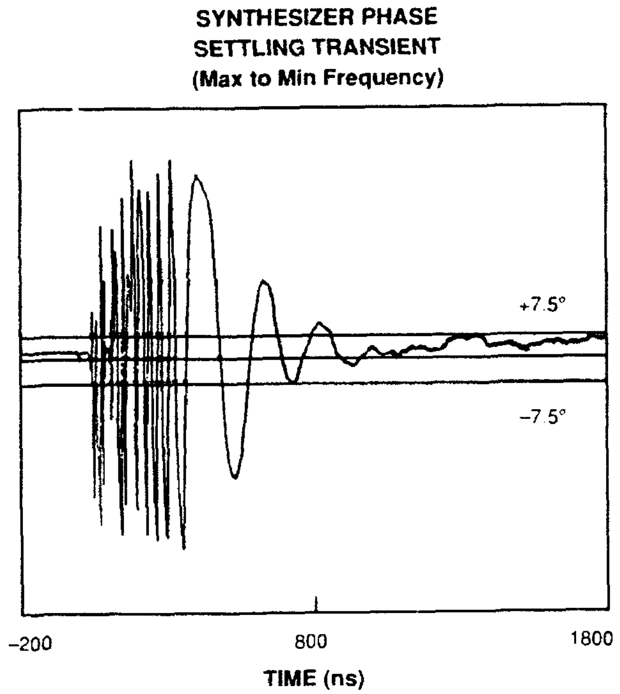
Figure 2. Frequency Synthesizer Reductions



- STRAIGHTFORWARD IMPLEMENTATION
- MMIC SWITCHES: SMALL/LOW POWER
- FILTER SIZE LIMITS N – LARGER DDS BW
  - HIGHER POWER DDS
  - LARGER SPURS

- LARGE N – SMALLER DDS BW
  - LOWER POWER DDS
  - SMALLER SPURS
- LARGE LOOP BANDWIDTH
- VERY HIGH SPEED COUNTER
  - POWER PENALTY
- CUSTOM VCO

Figure 3. Bandwidth Expansion Implementation Options



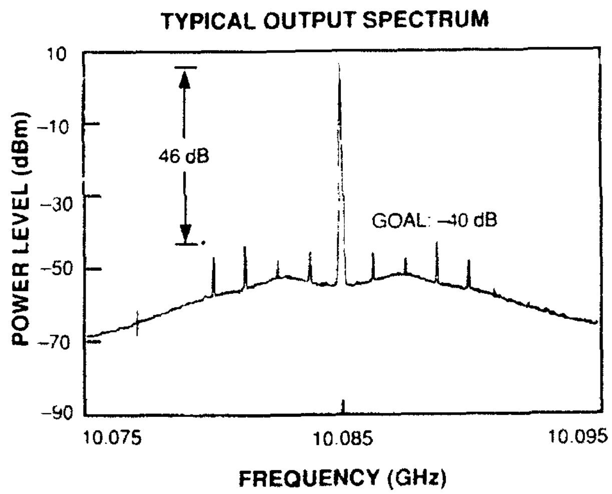


Figure 4. Breadboard Synthesizer Performance

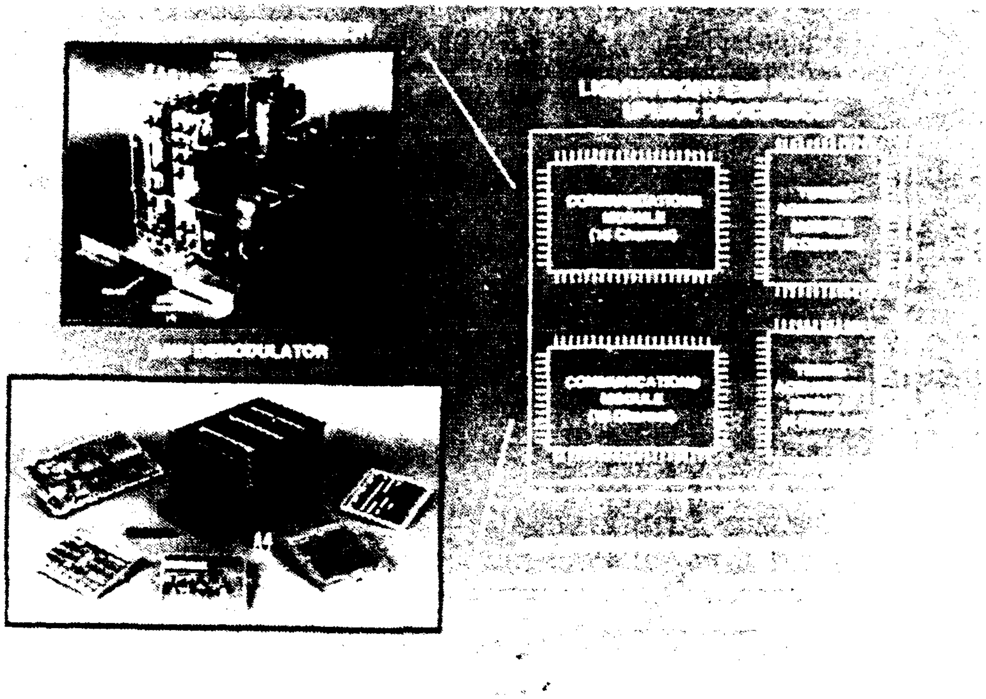
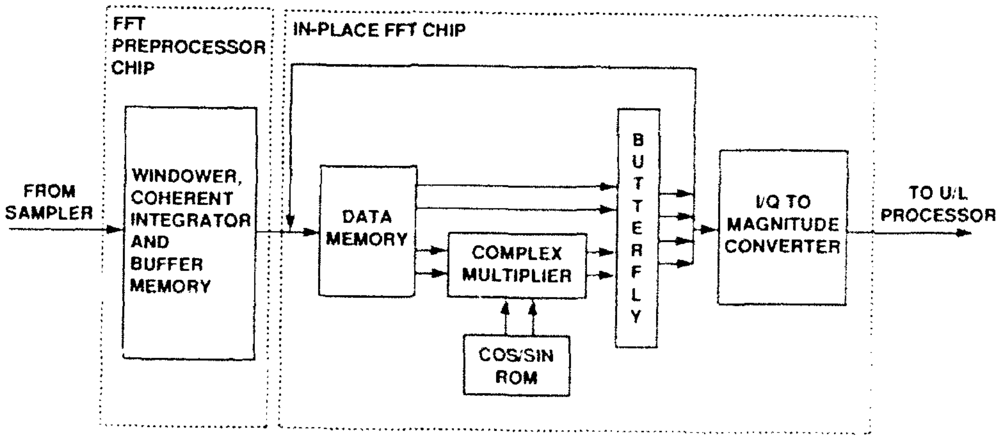


Figure 5. Lightweight LDR Signal Processor Implementation



- DEMODULATION FOR BOTH COMMUNICATIONS AND ACQUISITION
- LIGHTWEIGHT IMPLEMENTATION
- ALLOWS UP TO 256-POINT TRANSFORM

Figure 6. Digital FFT Demodulator

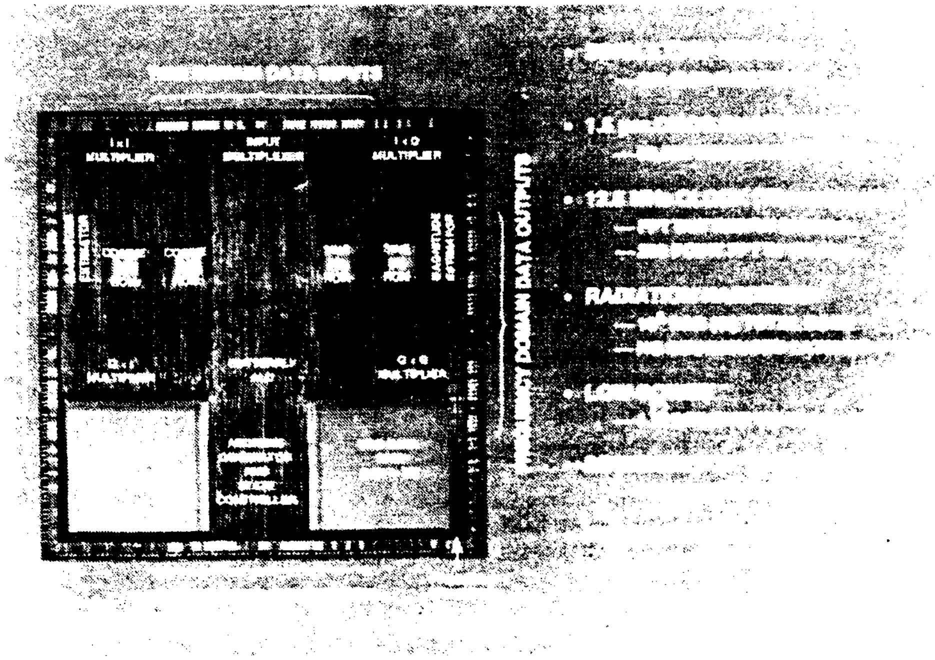


Figure 7. FFT Application Specific IC

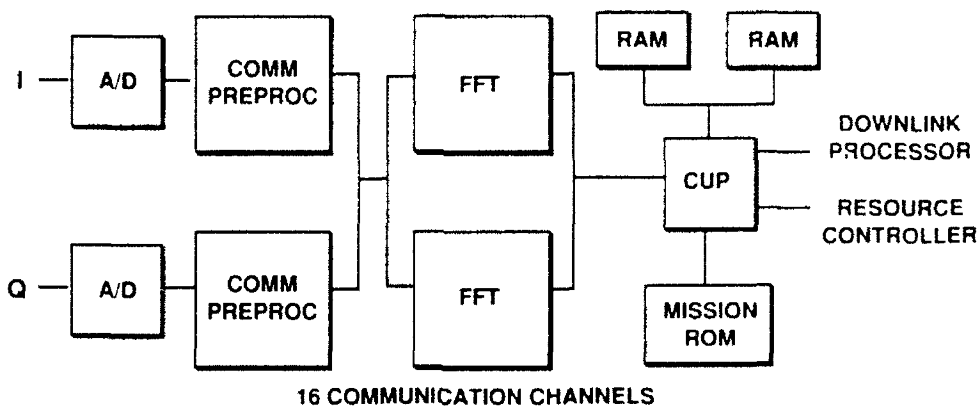
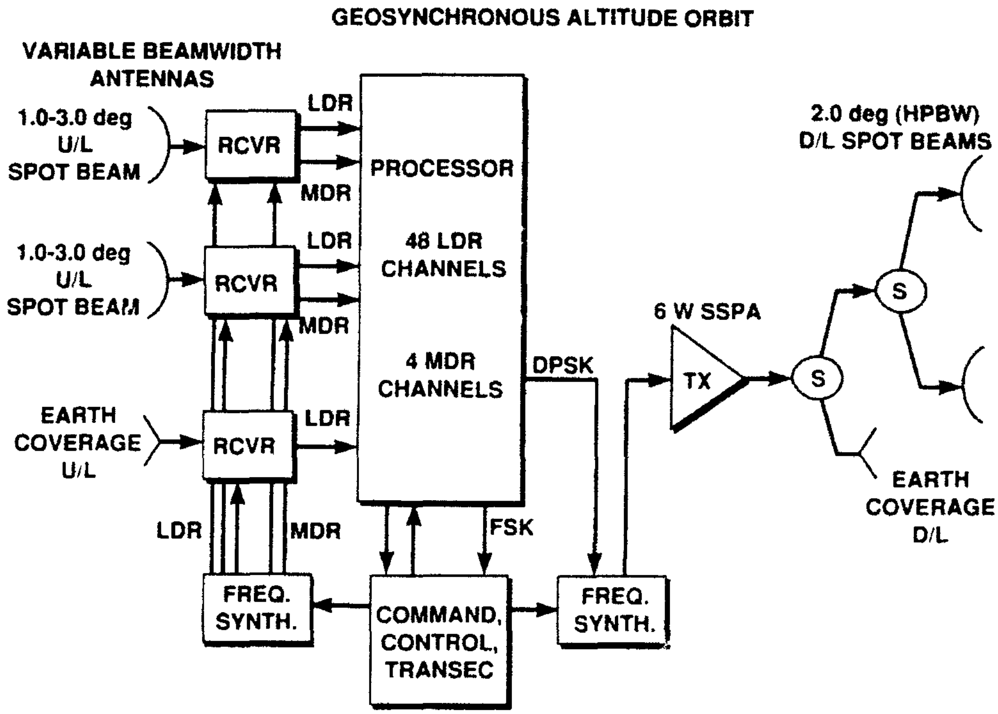


Figure 8. Demodulator and Uplink Processor Multi-Chip Module Configuration



PAYLOAD SIZING

EHF ONLY ~200 lb ~290 W

Figure 9. Example LDR/MDR Payload

EARTH COVERAGE BEAM

GROUND TERMINAL (6', 25 W) 1 × 2.4 kbps
GROUND TERMINAL (6', 25 W) 4 × 150 bps

3° SPOT BEAM

PORTABLE TERMINAL (2', 2 W) 4 × 2.4 kbps
PORTABLE TERMINAL (2', 2 W) 6 × 600 bps
TRANSPORTABLE TERMINAL (4', 12 W) 4 × 16 kbps
TRANSPORTABLE TERMINAL (4', 12 W) 4 × 2.4 kbps

1° SPOT BEAM

PORTABLE TERMINAL (2', 2 W) 8 × 2.4 kbps
TRANSPORTABLE TERMINAL (4', 12 W) 1 × 1.024 Mbps
TRANSPORTABLE TERMINAL (4', 12 W) 2 × 512 kbps
TRANSPORTABLE TERMINAL (4', 12 W) 4 × 256 kbps
TRANSPORTABLE TERMINAL (4', 12 W) 6 × 16 kbps

TOTALS

27 LDR NETWORKS, 45.0 kbps
17 MDR LINKS, 3.232 kbps

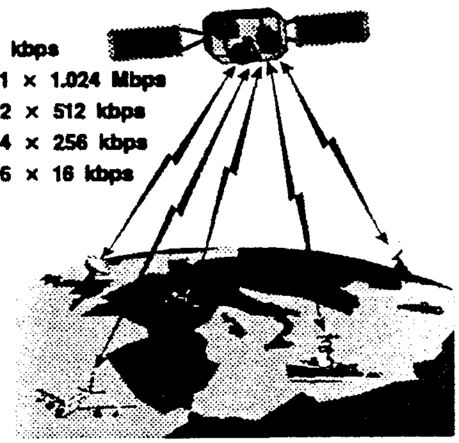


Figure 10. Example LDR/MDR Loading Scenario

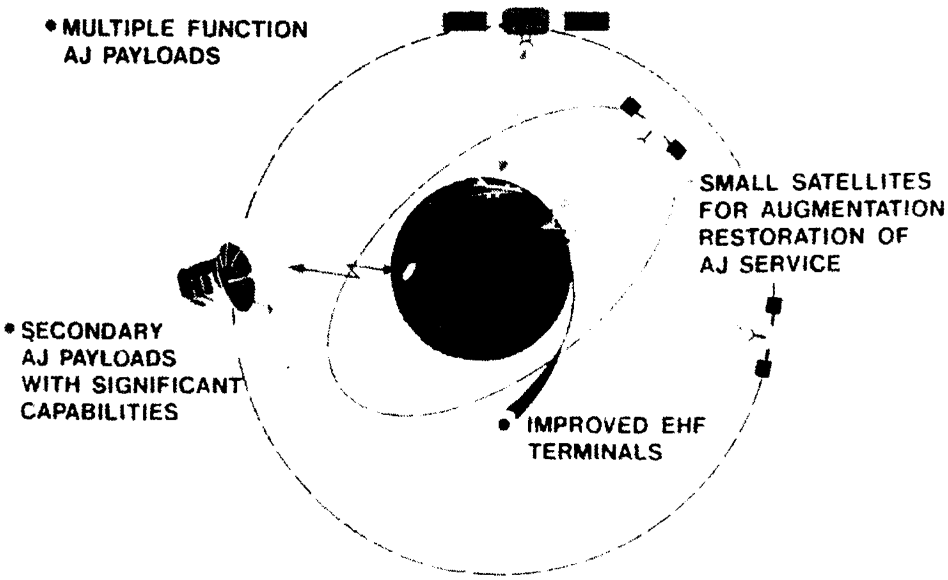


Figure 11. Technology Utility

SYNCHRONIZATION TECHNIQUES FOR MEDIUM DATA RATE EHF MILSATCOM SYSTEMS

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SUMMARY

In the area of MILSATCOM, considerable effort is currently being expended on the development of systems operating in the EHF frequency bands and employing onboard processing. Two of the principal advantages of employing these features are that the MILSATCOM system will be capable of increased throughput and will possess increased immunity to jamming and other channel impairments. Most of the previous research and development in EHF MILSATCOM has concentrated on Low Data Rate (LDR) waveforms, for data rates up to 2.4 kbps. In order to meet the requirements for increased throughput, Medium Data Rate (MDR) waveforms are being developed. These systems include data rates up to and including T1. This paper examines the constraints that communicating at MDR data rates places on synchronization and provides an overview of some of the various techniques and algorithms that can be employed for both spatial and time acquisition as well as tracking. Both channel and equipment impairments affecting synchronization are examined. Robust open and closed loop acquisition and tracking algorithms are examined in conjunction with onboard processing techniques. Performance is discussed in terms of SNR, acquisition time, probability of correct acquisition and probability of false acquisition. Tradeoffs in a MILSATCOM system design based on various user requirements are also presented.

SYMBOLS

dB	decibel
DL	downlink
EHF	Extremely High Frequency
E_h	hop energy
E_s	coded burst symbol energy
FDMA	Frequency Division Multiple Access
GHz	GigaHertz
GT	Ground Terminal
kbps	Kilobits per second
kHz	KiloHertz
L	diversity level
n_t	number of downlink TDMA slots
ns	nanoseconds
$P_D(DL)$	probability of detection on DL
$P_D(UL)$	probability of detection on UL
P_{DEMOD}	demodulator BER
P_{DIVS}	BER at output of diversity combiner
P_{DPSK}	BER of DPSK modulation
$P_{FA}(DL)$	probability of false alarm on DL
$P_{FA}(UL)$	probability of false alarm on UL
P_S	probability of overall synchronization
R_h	hop rate
R_s	coded burst symbol rate
TDMA	Time Division Multiple Access
TOD	Time of Day
T_{DL}	error in DL timing estimate
T_g	worst case group delay variation
T_{mtch}	UL hop timing mismatch
T_{UL}	residual error in UL timing
UL	uplink

1 INTRODUCTION

As is well known, utilizing the Extremely High Frequency (EHF) band for Military Satellite (MILSATCOM) communications offers several advantages over more conventional military satellite communications bands such as X-band or UHF. The EHF frequency bands centered at 45 GHz on the uplink and 20 GHz on the downlink (DL) allow the use of higher gain directed antenna beam patterns, which provides greater immunity to potential uplink jammers and allows a greater capacity on the downlink [1]. This is due to the narrower antenna beamwidths that can be attained at EHF frequencies, permitting the rejection of jammers either through null beamsteering or sidelobe rejection. These features are advantageous in a tactical scenario, in which satellite communications coverage is desired for a small theatre of operation. A second major feature that is desirable in a MILSATCOM system is on-board processing at the payload. This feature prevents uplink (UL) channel impairments such as fading or jamming from degrading the satellite transponder and degrading the downlink as well as the uplink. The majority of current research and development of EHF MILSATCOM systems has been for Low Data Rate systems supporting user data rates from 75 bps up to and including 2.4 kbps. Such data rates may be sufficient for limited voice or low capacity data communications within a theatre of operation, however, higher data rates supporting more robust voice communications and a higher volume of data traffic would be required for a theatre command center. For such scenarios, MDR type data rates up to and including T1 would be required.

Synchronization is one of the key areas of potential vulnerability in the design and operation of a MILSATCOM system. In order to meet operational requirements, an MDR EHF MILSATCOM system must provide a synchronization capability that is at least as robust as the communications capability it supports. Synchronization consists of both acquisition and tracking processes. These processes involve adjusting the critical parameters of spatial pointing, frequency and timing, at the ground terminal. It will be assumed that the payload is the system reference with regard to frequency and timing, and that the ground terminal must match its timing and frequency to the DL received signals. Similarly it must ensure that the UL signals arrive at the payload matched in time and frequency to those of the payload. This paper examines the system design considerations for synchronization of EHF MDR systems utilizing Tactical Satellites (TACSATS). The emphasis of the discussion will be on performance capabilities to meet theatre command requirements.

The paper is organized as follows. Section II examines some of the system characteristics and assumptions which impact the design of the synchronization process. Sections III, IV, and V examine the design of spatial, time and frequency synchronization algorithms, respectively. Section VI examines parameters and performance measures for an overall synchronization algorithm. Section VII provides a conclusion to the paper.

2 MDR EHF SYSTEM CHARACTERISTICS

2.1 Waveform and Operational Considerations

The assumed key waveform and operational characteristics that are pertinent to the discussion in this paper are outlined in Table I. These characteristics were considered by the Canadian Department of National Defence for the current MDR EHF MILSATCOM system design [2]. As discussed above, the uplink bandwidth is 2 GHz centered about 44.5 GHz, whereas the downlink bandwidth is 1 GHz centered about 21 GHz. Frequency hopping is employed to combat jamming, with the hopping rate being on the order of kHz. A typical system configuration is illustrated in Figure 1, in which a ground terminal (GT) in a given theatre of operation communicates with a GT in the same or different theatre. A theatre of operation is defined by the satellite antenna beam pattern coverage. The satellite antenna can either be a fixed or steerable spot beam, or a multi-beam antenna that can be dynamically steered from one theatre to another on a hop to hop or frame to frame basis. In order to accommodate multiusers and variable data rates, a hybrid FDMA/TDMA frame structure is employed on the uplink as illustrated in Figure 2. Users are assigned capacity in terms of an access which can be one or more nonconcurrent subframes in one or more user channels. The downlink employs a single channel TDMA structure. The modulation assumed for purposes of this discussion is DPSK. No pulse shaping has been assumed, although recent studies indicate that this may be useful for MDR applications [3]. M-ary FSK could also be considered, however given that slow frequency

Table I: MDR System Parameters

Feature	Specification
UL frequency	44.5 GHz
DL frequency	21 GHz
UL frequency bandwidth	2 GHz
DL frequency bandwidth	1 GHz
modulation	DPSK
Hopping rate	slow frequency hopping
Data Rates	32 kbps to T1
Processing functions	- coding: block or convolutional - diversity - interleaving - time permutation
satellite orbits	- arbitrary
payload antenna	2° - 5° beamwidth 26-34 dBi
payload amplifier	5 W SSPA
ground terminal antenna	up to 1.8 m
ground terminal amplifier	up to 20 W

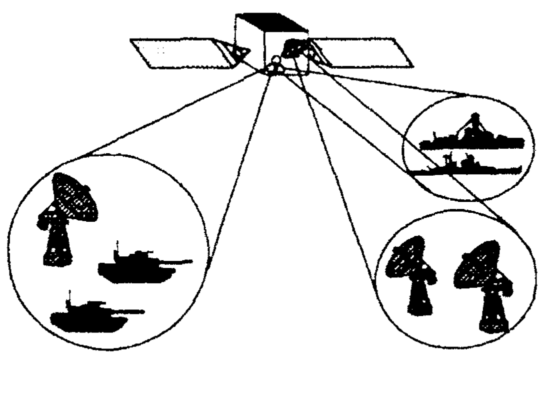

Figure 1: System Configuration

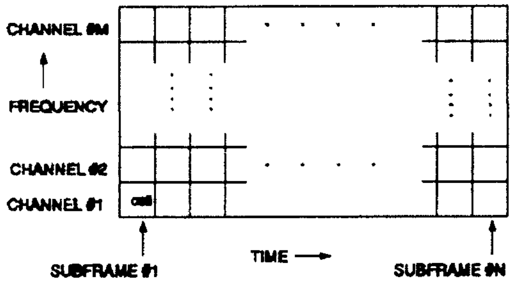

Figure 2: UL FDMA/TDMA frame structure

hopping is assumed (i.e. the hop rate is less than the data rate), a DPSK type modulation is more amenable to attaining the MDR data rates.

2.2 Impairments

In order to combat various channel impairments such as additive noise, fading, and scintillation as well as jamming threats it is assumed that communication data processing techniques such as FEC coding, interleaving, diversity and time permutation of data hops are implemented in the system. In addition to the above channel and ECM threats the system must be capable of operating under system and equipment induced impairments such as doppler effects due to the relative motion between the payload and the GT, relative payload to GT clock drift, RF group delay variation, and RF frequency response or the hop-to-hop amplitude variation as a function of frequency. These impairments are listed in Table II along with the assumed values for the purposes of discussion in this paper. The doppler effect is negligible for geostationary orbits, however for non-geostationary, elliptical type orbits such as Tundra or Molniya, the doppler effects can be as severe as documented in Table II. These orbits would be required in a tactical theatre of operation at northern latitudes, particularly in the Arctic.

2.3 Link Budget Requirements

For a TACSAT EHF system, it is assumed that a 5W SSPA is employed on the payload. Larger amplifiers would require the use of a TWTA, which would impact on the available size and weight of the payload. The antenna is assumed to be a spot beam with a 5° beamwidth. For this size of amplifier on the payload, typical receive C/N_0 levels at the ground terminal will vary from 50 to 60 dB-Hz for a geostationary orbit in clear sky conditions. It should be noted that the link budget must also accommodate rain fade margins on the UL and the DL. Depending on the desired fade margin this could potentially restrict the DL burst data rates under given channel conditions.

2.4 Initial Acquisition Estimates

The complexity of the acquisition phase of the synchronization process will depend significantly on the

Table II: MDR System Impairments

Impairment	Magnitude
Additive Noise	channel dependent
Scintillation	channel dependent
Doppler	< '90 kHz
Clock Drift	1 x 10^{-9}
Group Delay Variation	40 ns
Frequency Response Variation	+/- 3 dB
Jamming	channel dependent

magnitude of the errors in the initial GT estimates of payload pointing, frequency and timing.

2.4.1 Timing Estimate

The error in the initial time estimate is determined by the availability of an accurate time reference and the relative rate of drift of the time standard maintained by the GT with respect to the reference time standard in the payload. With the availability of single board GPS receivers, an MDR terminal can easily maintain time accuracies on the order of microseconds. If GPS or similar time reference systems are not available, the initial error in the estimate of the time error would likely be on the order of milliseconds. Given the current level of technology, an MDR terminal can easily accommodate a rubidium standard. A typical value of relative drift rate for a rubidium standard is 5 x 10^{-12} per day. The resulting error, even after several days of uncalibrated use would still be less than a microsecond. For purposes of discussion, an initial timing error is 50 milliseconds is assumed.

2.4.2 Spatial Pointing Estimates

The accuracy of the initial spatial pointing estimate is dependent on three principal parameters; namely, the error in the GT's estimate of its own location, the initial timing estimate, and the accuracy of the GT's ephemeris algorithm. The dependency on the time estimate results from the fact that the GT's estimate of the satellite position is an output of the ephemeris algorithm, which employs the current time estimate as an input. If GPS is available, both the initial time estimate, as well as the GT's position will be known to a high degree of accuracy. Furthermore, if the MDR base station is fixed in a tactical theatre of operation, the location of the GT will typically be known to within 10's of meters, either from survey or navigational aids and an accurate time standard can be safely assumed. In either of the scenarios described above, the only effective error will be due to the accuracy of the ephemeris algorithm. A very conservative estimate is that the error component in spatial pointing due to the ephemeris algorithm will be known to within 0.1° in which case spatial acquisition may not be necessary. If GPS or similar time standards are not available and the location of the GT is not precisely known, the contribution of the spatial pointing error due to the error in the estimate of the GT location will become significant. In a tactical field scenario the GT location will typically be known to much better than 10 km., resulting in a contribution to the spatial pointing error of less than 0.1°. Thus the total pointing error should be less than 0.2°.

2.4.3 Frequency Estimates

The initial error in the frequency estimate will be due to a combination of the accuracy of the ephemeris algorithm in estimating motion induced doppler effects and the relative frequency drift between the time standards of the payload and the ground terminal. For the orbits referenced in section 2.2, the worst case unresolved doppler error will be on the order of 1 to 2 kHz. Again assuming the presence of a rubidium standard in the GT, the frequency error due to relative clock drift will be negligible for purposes of .hese discussions.

3 SPATIAL ACQUISITION

To achieve spatial synchronization, the ground terminal must be capable of acquiring and tracking the pointing of the antenna typically to within an angle corresponding to a 1.0 dB loss in gain. In an MDR system there are several factors that will impact on the spatial synchronization algorithm to be selected. If the initial spatial pointing error is relatively large, such as on the order of one to two 3 dB beamwidths, spatial acquisition will likely have to consist of two phases, a coarse spatial acquisition phase, and a fine spatial acquisition phase. The coarse spatial acquisition phase can consist of a stepped or continuous search pattern, normally in steps of one 3 dB beamwidth, out from the best initial estimate as provided by the ephemeris algorithm.

Since in MDR the DL is capacity limited, it is desirable to have an antenna with as high gain as practical. However high gain antennas have narrower beamwidths, resulting in larger attenuations for the same error in spatial pointing. Typical UL and DL antenna beam characteristics for EHF frequencies are given in Table III as a function of antenna diameter. As discussed in section 2.4.2, the initial pointing error for an MDR terminal is usually less than 0.1°, in which case a coarse spatial search stage is not required. For example, it can be seen that for a 2.4 meter antenna, that a pointing error of 0.1° corresponds to the 3 dB beamwidth. For a tactical environment in which larger initial pointing errors may be incurred, the antenna size will typically be smaller, on the order of 1 meter. In such a scenario the pointing error of 0.2° discussed in 2.4.2 will fall within the 3 dB beamwidth. In such cases spatial acquisition would only need to consist of the fine spatial acquisition stage. The most common algorithm employed is a conical scan tracking loop about a 1 dB contour. Such loops can rapidly acquire in AWGN in the order of seconds from initial pointing errors within the 3 dB contour of the antenna. This is sufficient for most MDR EHF antennas as discussed above.

Table III: Typical EHF Antenna Characteristics

Antenna Diameter	1 dB Point (degrees)	3 dB Point (degrees)
0.14 DL	2.2	3.8
0.14 UL	1.0	1.75
1.0 DL	0.3	0.53
1.0 UL	0.14	0.24
1.8 DL	0.18	0.3
1.8 UL	0.09	0.18
2.4 DL	0.13	0.21
2.4 UL	0.06	0.1

If the initial pointing error exceeds the 3 dB contour of the antenna, or if jamming or scintillation is present, it may be necessary to employ a coarse acquisition search. Spiral or stepped searches may be employed, however there is an additional consideration for an MDR terminal. If the pointing error is outside the 3 dB region, it is possible that the spatial search may point to a sidelobe at the payload. In a high SNR environment a detection may occur, resulting in a false locking of the spatial pointing on a sidelobe. One solution is to collect statistics from all possible search regions, however this impacts the acquisition time. An alternate approach is to employ a gimbal scan about a larger contour than that employed for the conical scan. The processing would be identical. If the contour of the gimbal scan is judiciously chosen with respect to the overall search region, the gimbal scan will always encompass the mainlobe of the antenna. Other considerations include the increased sensitivity of the conical scan to pointing errors due to the higher slope of the beamshape in high gain antennas, as well as the effects of frequency flatness variation on the stability of the conical scan processing.

4 TIME ACQUISITION

4.1 Overview

Time acquisition is divided into two stages; namely, DL and UL time acquisition. Due to the tactical requirement that the MDR EHF system have a multiple access capability, it is necessary for the payload to dynamically assign data hops and theatres. Since a GT in a given theatre does not have knowledge of which hops have actual data until after acquisition has been completed, DL synchronization is accomplished through the use of specialized synchronization hops. These hops must occur in known locations in the downlink datalink structure. DL acquisition is achieved by detecting the presence of the synchronization hops which are periodically transmitted to each theatre of operation. Since the detection of these hops cannot take place unless the antenna is pointing at the payload, the time acquisition process is performed concurrently with the spatial acquisition procedure described in section 3. As for spatial acquisition, the DL acquisition process consists of a coarse and fine stage. The coarse stage synchronizes the DL timing to within 1 DL hop, providing the ground terminal knowledge of the correct point in the frequency hopping sequence relative to the GT time-of-day (TOD). A second fine stage of DL time acquisition is required to attain the DL timing to within a DPSK chip or symbol period.

It should be noted that the use of time permutation to combat partial time jamming adds complexity and delay to the acquisition process. The position of the permuted synchronization hops must be calculated based on the estimated TOD and the hopping pattern. The additional delay will be on the order of the period over which the permutation takes place, for example a frame.

4.2 Coarse DL Time Acquisition

Since the DL is in a TDMA format, the SNR in a hop is related to the required symbol SNR by the following relationship

$$\frac{E_h}{N_0} = \frac{E_s}{N_0} \cdot \frac{R_s \cdot n_c}{R_h} \qquad (1)$$

in which E_h/N_0 is the hop-energy-to-noise-spectral-density ratio, E_s/N_0 is the coded-symbol-energy-to-noise-spectral-density ratio, R_s is coded symbol rate of a single channel, n_c is the number of downlink TDMA slots corresponding to the number of uplink channels, and R_h is the hop rate. For data communications the specified value of E_s/N_0 is fixed by the required bit error rate, and the hop rate is typically fixed for a given system. Thus as the number of uplink channels and/or the channel data rate is increased, the level of E_h/N_0 must be increased to maintain the same level of E_s/N_0. Since MDR systems must support data rates up to T1, the resulting E_h/N_0 is high. As an example suppose a MDR system has 8 channels, rate 1/2 coding, a hop rate of 10 kHz and a T1 data rate. For a typical required system error rate between 10^{-5} and 10^{-6}, the required value of E_s/N_0 is roughly 7 dB. For the values noted above, the resulting E_h/N_0 will exceed 40 dB. Similar values can be obtained for a jamming environment. Detection of the coarse DL synchronization hops can thus easily be made based on energy measurements. The format of the coarse synchronization hops need not be complex, but should be selected to allow a high probability of detection (P_D), and a low probability of false alarm (P_{FA}). This is a common radar problem, and numerous waveforms can be selected[4]. A simple example is a modulated CW tone.

Fine DL time acquisition is necessary to obtain the DL timing to within a symbol period, as well as to recover the symbol rate clock. The DL timing accuracy is affected by the following factors; satellite motion induced doppler effects, relative clock drift between the payload reference clock and the GT clock, group delay variation introduced by the channel or the equipment, and the timing jitter introduced by the synchronization process itself. As discussed in section 2.4.3, the unresolved doppler can be up to 2 kHz, resulting in a timing variation on the order of 100 nsec/sec. Furthermore, the group delay variation can be on the order of 40 nsec. Given the TDMA nature of the MDR downlink, and data rates up to T1, the burst symbol rate of the DL will typically be on the order of 10 MHz to 100 MHz, depending on the number of slots. In the example given above the symbol period would be 40 nsec. Fine timing on the DL can be achieved by sending fine synchronization hops consisting of a unique word which can be correlated at the GT. The number of fine synchronization hops and the period between arrival of fine synchronization hops is dictated by magnitude of the timing error that accumulates between synchronization hops. The number of DL synchronization hops must be sufficient to allow for all theatres of operation to be visited. The principal source of timing error that will be removed by the fine synchronization hops is the unresolved doppler error as well as clock drift error. Once fine timing has been obtained on the DL, the GT must track the DL timing. Doppler and clock drift must be tracked in order that the DL and UL timing of the GT can be adjusted to compensate for these effects. The timing jitter induced by the synchronization algorithm is a function of the complexity of the feedback loop design and can be separately minimized.

4.3 DL Time Tracking

In addition to tracking doppler and clock drift, each DL data hop must be compensated for the effects of group delay variation. At the MDR burst symbol rate, the group delay variation from one hop to the next can be one or more symbol periods as discussed in section 4.2. Thus each data hop must be resynchronized to the DL timing to ensure that no data symbols are lost or that the symbol timing of the received symbols is not misaligned. This can be achieved by encoding a known preamble at the start of each data hop. The GT will correlate the received preamble against the transmitted preamble and align the timing to within half a symbol period by detecting the peak of the correlation process. The length of the unique word selected for the correlation process is a function of the SNR that is required to obtain the desired P_D and P_{PA} of each data hop. It should be noted that these values do not have to be overly stringent, since in fact the probability of synchronizing P_S is driven by the required BER performance [5]. If the preamble in a given hop is not properly correlated, on average half of the symbols in the hop will be in error. Thus the probability of error out of the GT demodulator is given by the following expression

$$P_{DEMOD} = P_{DPSK} \cdot P_S + 0.5\ (\ 1 - P_S\ ) \qquad (2)$$

in which P_{DEMOD} is the probability of bit error at the output of the demodulator, and P_{DPSK} is the probability of bit error of a DPSK modulated signal. If the demodulator is followed by diversity combining, the value of P_S can be poor, and yet still allow robust performance. For example assuming a basic majority logic combining approach, in which hops are diversified the BER at the output of the diversity combination circuitry will be given by

$$P_{DIVS} = \sum_{i=L/2}^{L} \binom{L}{i} P^{i}_{DEMOD} \cdot (\ 1 - P_{DEMOD}\ )^{L-i} \qquad (3)$$

in which P_{DIV} is the probability of bit error at the output of the diversity combination circuitry. Plots of P_{DIVS} versus E_s/N_0 are given in Figure 3. It can be seen that even with a modest diversity of L = 11, that a BER of less than 0.02 can be obtained at low E_b/N_0 even for P_S values as poor as 0.7. This BER into a good FEC will give an overall BER of less than 10^{-5}. In an operational system, other more robust diversity combining techniques may be employed.

4.4 Unique Word Selection

The probability of false alarm in the detection of an erroneous preamble is dictated by the size of the sidelobes of the unique word. Unique words with ideal correlation properties such as Barker type codes can only be found for a few shorter length sequences, where ideal correlation of a unique word of length M shall is defined as M for perfect alignment and +/- 1 elsewhere. By shortening the decision window of the unique word and considering only sidelobes within the defined window, it is possible to attain a higher processing gain for the unique word. The width of the window must be long enough however, to accommodate any potential group delay variation and accumulated clock drift.

4.5 UL Time Acquisition and Tracking

When the GT initially begins acquiring, the total time uncertainty is dominated by the error in the estimate of the range to the satellite and the error in the position of the GT. Acquiring the DL timing resolves one of these parameters. In order to resolve the second parameter, the time at which the UL transmissions occur must be adjusted to ensure that the data hops arrive at the payload within the processing window for the UL hops. This is best achieved by a closed loop process in which specialized synchronization probes are transmitted in known waveform locations to the payload. The payload detects the position of these hops relative to the nominal locations and transmits specialized responses to the GT that contain the required adjustment of the GT UL timing in order to ensure that the UL hops transmitted by the GT arrive at the payload with the correct timing. As is the case for the DL timing, the UL probes will contain unique words that should be judiciously chosen to allow for an optimal value of P_D and P_{PA}, relative to the size of the search window for UL timing.

After UL time synchronization has been achieved, the accuracy of the UL hop period matching is constrained to

$$T_{match} \leq \pm\ (\ 2T_{DL} + T_g + T_{UL}\ ) \qquad (4)$$

in which T_{match} is the UL timing hop mismatch, T_{DL} is the error in the DL timing estimate, T_g is the worst case group delay variation, and T_{UL} is the residual error in the UL timing from the previous UL time estimate. The residual UL error T_{UL} is no worse than the accuracy provided by the coarse or fine DL synchronization responses. For MDR systems, T_{match} will exceed the period of a DL symbol, requiring that the UL data hops have preambles with unique words to permit hop-to-hop correlation of the timing. The size of T_{match} will dictate the size of the processing window in the payload and the size of guard bands in the UL hops.

An additional consideration in the design of an UL timing and tracking protocol is the placement of the probes in the UL frame structure. Probes can be assigned on either a channel basis over the entire frame, or on a TDMA time slot basis, spanning some or all of the available channels. Both of these approaches have advantages and disadvantages. UL probes assigned on a channel basis will provide more throughput to an assigned user; however, the payload must possess separate front-end processing to detect probes from one theatre of operation while it processes data channels from a separate theatre of operation. In contrast, if probes are assigned on a TDMA time slot basis, the payload can detect probes from one theatre while receiving data communications from another theatre all within the same frame, with no additional hardware. The disadvantage to this approach is that the amount of probe capacity available to the GT is more restricted than in the FDMA approach.

5 FREQUENCY ACQUISITION

The principal degradation in system performance due to frequency error will be due to the accuracy of the bit timing in the DPSK demodulator of the MDR receiver. In an MDR system, the coded symbol rate will be greater than twice the T1 rate, or 3.2 MHz, for rate 1/2 FEC coding. Based on a relative frequency error on the order of 1 to 2 kHz, the resulting normalized bit timing error at the receiver will be less than $\Delta fT = 0.001$. The resulting degradation in a DPSK modulator for this range of bit timing error is negligible. Thus frequency acquisition and tracking is not required for an MDR system.

6 SYSTEM SYNCHRONIZATION ALGORITHM

In a tactical environment it is paramount that the overall synchronization time be kept to a minimum while simultaneously providing capacity to synchronize many users at once. This goal can be accomplished through a combination of judicious design of the initial parameter estimation capabilities of the GT, selection of a robust synchronization algorithm and optimization of the parameters of the synchronization algorithm affecting synchronization time. The availability at the ground terminal of accurate TOD information, and an accurate ephemeris algorithm will greatly reduce the spatial-temporal search region, allowing rapid acquisition. However the algorithm must be robust in the sense that if accurate initial estimates are not available, the GT will still acquire the payload. In addition to minimizing the acquisition time, the algorithm must also simultaneously minimize the probability of false synchronization. Considering the DL and UL acquisition processes separately, the overall probability of correct synchronization and false synchronization are given by

$$P_D = P_D(DL) \cdot P_D(UL) \tag{5}$$

and

$$P_{FA} = P_{FA}(DL) \cdot P_D(UL) + P_D(DL) \cdot P_{FA}(UL) + P_{FA}(DL) \cdot P_{FA}(UL) \tag{6}$$

in which $P_D(DL)$, $P_D(UL)$, $P_{FA}(DL)$, and $P_{FA}(UL)$ are the probabilities of detection and false alarm for the DL and UL synchronization processes. Figure 4 illustrates a simplified state diagram of the overall synchronization process. The design of the synchronization algorithm involves choosing the values of $P_D(DL)$, $P_D(UL)$, $P_{FA}(DL)$, and $P_{FA}(UL)$ so as to minimize the acquisition time under the desired tactical operating conditions.

For the overall probability of detection defined in equation (5), the detection requirements can be equally partitioned to the UL and the DL as a starting point in the design. However, it may be desirable to have a higher P_D on the DL and a lower P_D on the UL under certain operating conditions. A DL receive only mode of operation is an example of such a scenario. The DL detection process itself is composed of the coarse and fine spatial and time acquisition processes described in sections 4 and 5 above. Thus $P_D(DL)$ can be partitioned into a function of the values of P_D and P_{FA} for the individual coarse and fine synchronization processes. These values are a function of the number of detected synchronization hops that constitute an overall detection (i.e. a choose m out of n detection criteria), the SNR at which they are received, and the channel conditions. Due to the spatial search algorithm and the frequency flatness variation over the hopping frequencies, the received carrier power itself will vary with frequency and spatial pointing. The flatness variation results in a uniformly distributed received signal level with frequency, whereas spatially, the received signal level is a function of the antenna beamshape and the size of the spatial search region.

In equation (6), the first term can be neglected, since it is extremely unlikely that the UL will successfully acquire if the DL is not properly synchronized. The dominant term in the overall process is the second term, which is the probability of false synchronization of UL timing given the DL is correctly acquired. The third term is small in comparison to the second term if $P_{FA}(DL)$ and $P_{FA}(UL)$ are reasonably chosen. It should be noted that even though the first term is negligible, a large value of $P_{FA}(DL)$ will significantly impact the acquisition time, since the DL acquisition process will be dependent on the UL process to detect false alarms, which is obviously a poor design.

In order to be robust in the presence of jamming, the coarse and fine sync detection processing will require a verification stage. The verification stage in general will be an m choose n type of decision, in which m of n expected sync hops will have to be detected in order for the given stage of the detection process to be declared valid. It should be noted that a high P_D and low P_{FA} for the system does not necessarily imply a requirement for high a P_D and low P_{FA} on individual coarse or fine DL sync hops. For example, if the system requires rapid acquisition, then the values of P_D and P_{FA} for the individual sync hops must be close to the P_D of the overall system, to ensure that there are few false detections, otherwise the synchronization process will expend time verifying false detections. This would require m and n chosen to be large. Such an implementation would limit the number of sync hops that the system could support relative to its data capacity, or limit the number of theatres that the payload could provide synchronization hops to. On the other hand the overall P_D and P_{FA} could be met by lower values of m and n, which would result in more false alarms and verification steps, and potentially a longer acquisition time. The design chosen will depend not only on the capacity of the datalink structure to support the algorithm, but the size of the terminals. Acquisition by larger terminals in clear sky conditions can be supported by an algorithm with low m of n, however smaller terminals will require longer acquisition times, particularly in a TACSAT environment in which the DL SNR is limited.

7 CONCLUSION

This paper has examined the system design considerations for synchronization of an EHF MDR system employing a

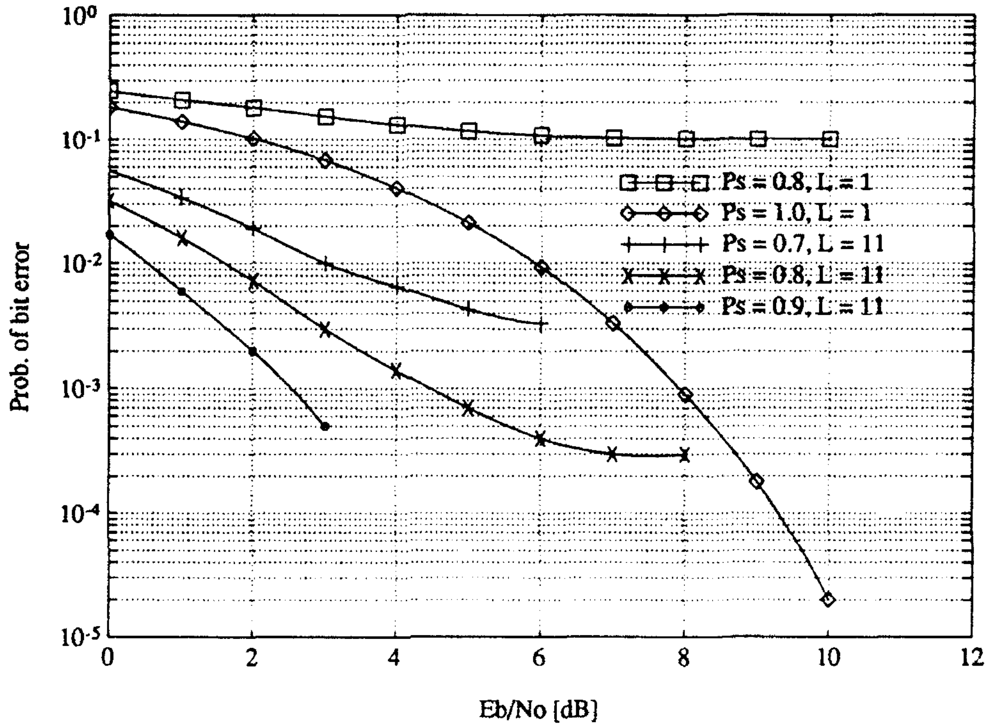

Figure 3: BER as a function of P_s

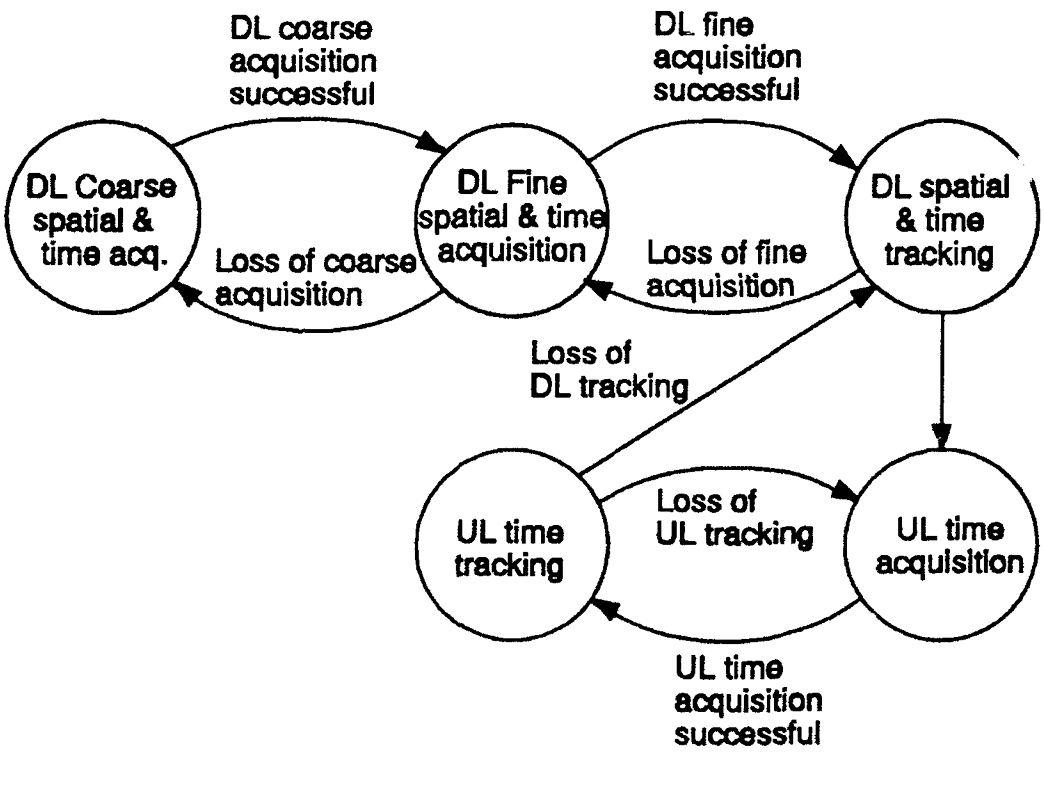

Figure 4: Simplified Synchronization State Diagram

TACSAT payload. Spatial and time synchronization algorithms have been presented and the tradeoffs in the overall acquisition and synchronization performance have been discussed. Based on the tactical requirements of the theatre of operation, the system designer must judiciously choose the parameters of the algorithm to ensure robust operation while simultaneously minimizing the acquisition time.

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Terrain & Coverage Prediction Analyses for Non-Geostationary Orbit EHF Satellite Communications

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1 SUMMARY

This paper examines communications coverage from five non-geostationary satellite orbits. These orbits are circular inclined synchronous, GPS, Molniya 12 and 24 hour, and Tundra orbit. It also shows the increase in propagation loss due to irregular terrain and the foliage loss at mm waves, as well as the combined effect of terrain and foliage on satellite view duration. It concludes that coverage calculations for EHF satcom earth terminals need to take into account terrain and foliage losses.

2 LIST OF SYMBOLS

°	degree
'	minute
AMSL	above mean sea level
CIS	circular inclined synchronous
ft	feet
GPS	global positioning system
kft	kilo-feet (1000 ft = 304.8 m)
km	kilometre
mi	mile (5280 ft = 1609.34 m)
Mol12	Molniya 12 hour orbit
Mol24	Molniya 24 hour orbit
Tundr	Tundra orbit
WA	Washington State, USA

3 INTRODUCTION

Geostationary satellites are expensive, afford poor visibility and degraded performance in the northern zones and may be not have coverage over a specific theater. Lightweight satellites tactical satellites (TACSATS) in non-geostationary orbits for use over a specific theater are an alternative.

Satellite "footprint" estimation invariably ignores the effect of terrain on coverage and communications[1-6,8,11-16]. This is justifiable when the earth terminal is in a barren flat terrain, and the satellite elevation angle is high. However, there may be considerable effect of the surrounding irregular terrain (mountains), in particular when the satellite elevation angle is small, on communications with a manpack, a shipborne or a transportable terminal. This paper will examine the effect of terrain and foliage on communications coverage of earth terminals with non-geostationary satellites.

For tactical reasons, there is a need for communication that is available continuously, is secure, has low probability of intercept, has nuclear survivability, is immune to EMP, has anti-jam features, and is *reliable*. Reliability is determined by *system availability* (99.99% etc.) and is defined by the system specifications. Although atmospheric absorption loss, effect of rain, nuclear event, jamming, etc. are included in the "fade" margin, the effects of terrain and vegetation are usually ignored. Inclusion of terrain, obstruction and vegetation losses will make the system specifications more complete, so these losses are examined in this paper.

4 NON-GEOSTATIONARY ORBITS

The TACSATS may use polar orbits, inclined circular orbits (e.g. circular inclined synchronous), and inclined elliptical orbit (e.g. Molniya and Tundra). The elevation angle and its variation is determined by orbit and location of the earth terminal. The elevation and azimuth and its variation with time for the above orbits are presented for Seal Rock, Washington, USA.

4.1 Orbits

The five non-geostationary orbits considered here are listed below:

* *Circular Inclined Synchronous (CIS)*
* *Global Positioning System (GPS)*
* *Molniya 12 Hour (Mol12)*
* *Molniya 24 Hour (Mol24)*
* *Tundra (Tundr)*

To determine the size, shape, orientation relative to earth of an inclined orbit, and the position of a satellite in its orbit, 6 orbital parameters are required. For the five orbits listed above these 6 parameters are given in Table 1.

4.2 Elevation Variation

Variation of elevation over a 24 hour period of the satellites in different orbits as viewed from Seal Rock Washington State, USA are shown in Fig. 1. The Molniy 24 hour orbit rises to an impressive elevation of approximately 89°. The maximum elevation angle is 36°

TABLE 1
ORBITAL PARAMETERS

Parameters	Values for Circular Inclined Synchronous Orbit	Values for GPS-012 Orbit	Values for Molniya 12 Orbit	Values for Molniya 24 Orbit	Values for Tundra Orbit
Catalog Number	cis	gps-012	mol12	mol24	tundr
Epoch Time	78/06/20 21:36:00 UTC	89/02/25 01:41:49 UTC	78/06/20 21:36:00 UTC	78/06/20 21:36:00 UTC	78/06/20 21:36:00 UTC
Element Set	1	2	3	4	5
Inclination	40.0°	55.1294°	63.4°	63.4°	63.4°
RA of Node	345°	216.2430°	275°	65°	20°
Eccentricity (e)	0.0	0.0090905	0.73	0.73	0.374
Arg of Perigee	-90°	180.3094°	-90°	-90°	-90°
Mean Anomaly	0.0°	179.6527°	0.0°	0.0°	0.0°
Mean Motion *(Rev/Day)*	One	2.01388764	Two	One	One
Decay Rate, Drag, *(Rev/Day²)*	0.0e+00	1.500e-07	0.0e+00	0.0e+00	0.0e+00
Epoch Rev	One	17	One	One	One
Semi-Major Axis (a)	42164 km	26487.80 km	26561.789 km	42164 km	42164 km

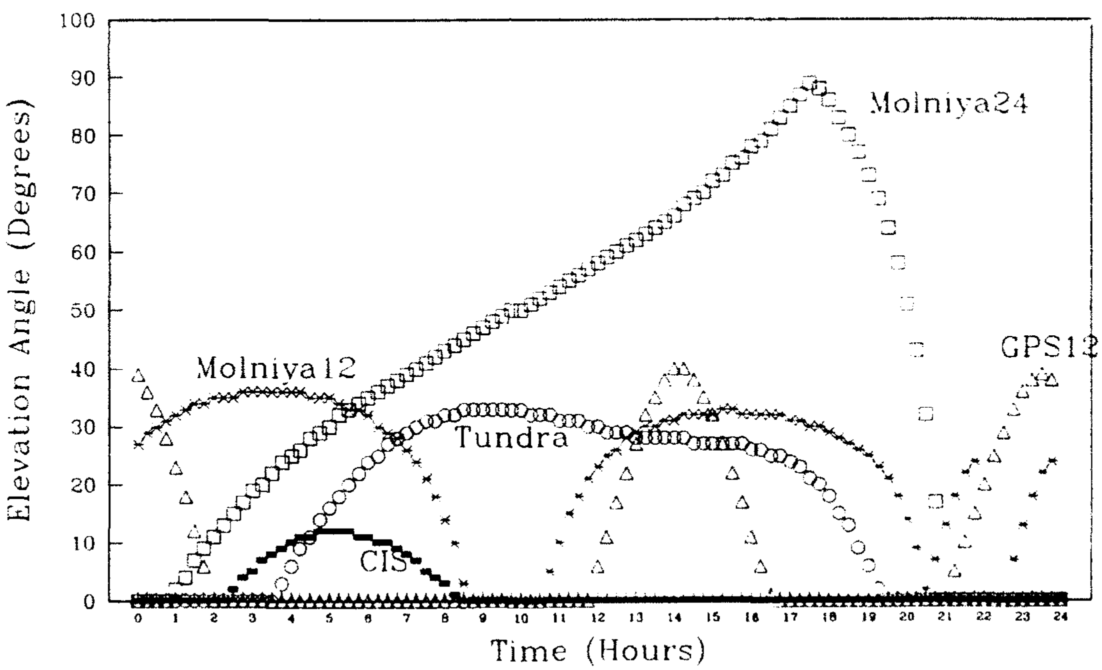

Fig. 1. Elevation angle variation for different orbits.

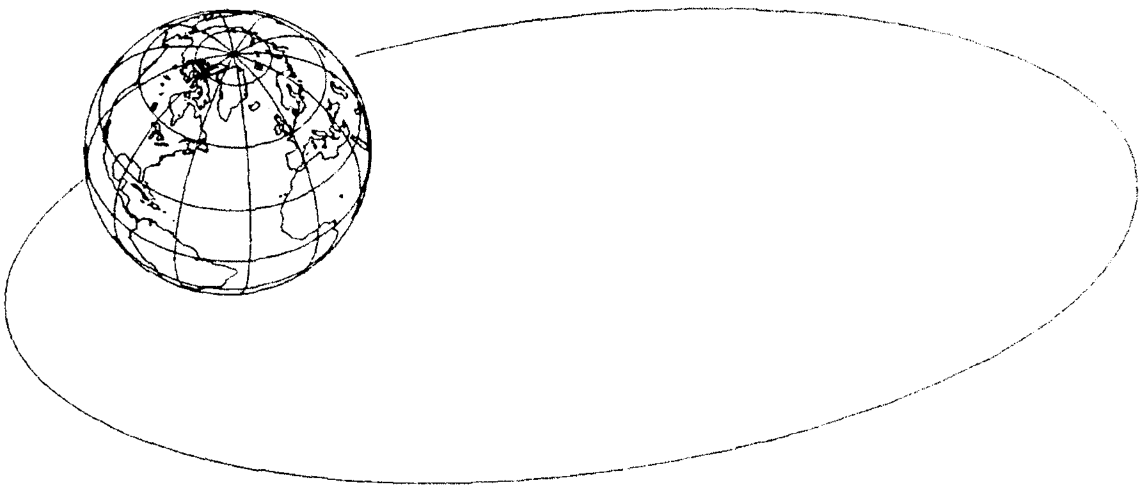

Fig. 2. The Tundra orbit has an eccentricity of 0.374
and inclination of 63.4 degrees.

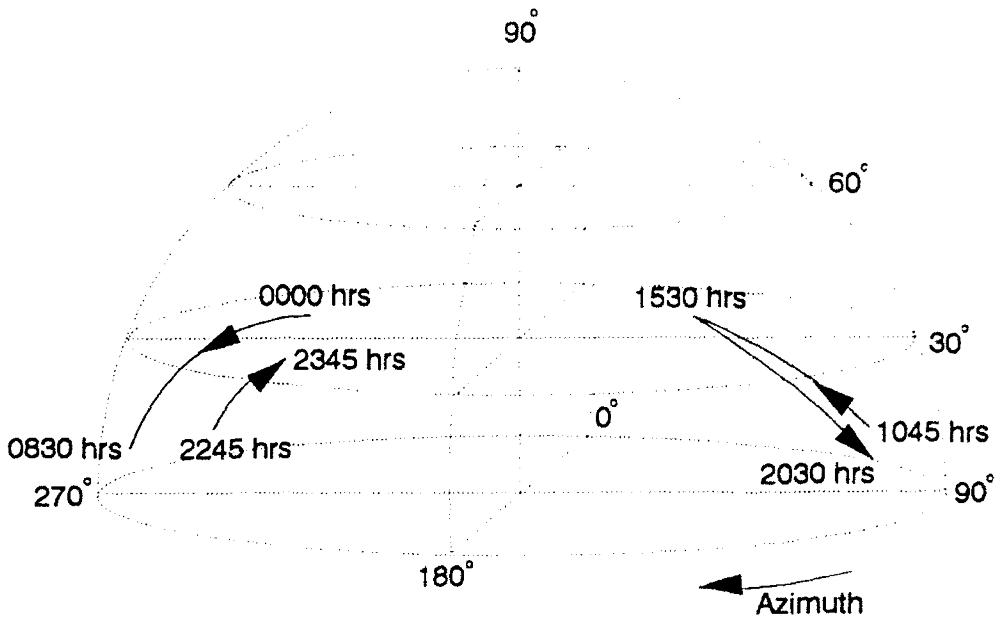

Fig. 3. Azimuth and elevation variation for the Molniya 12 orbit.

for Molniya 12 hour orbit, 33° for the Tundra orbit, 39° for the GPS orbit, and 12° for CIS orbit. Of the five orbits Molniya 24 and Tundra are particularly attractive because elevation angle is > ≈30° for 12 hours.

The Molniya and Tundra orbits are known for their high degree of eccentricity 0.73 and 0.374 respectively, and an inclination (63.4°). Their key benefits are coverage of the northern latitudes and a prolonged hang-time at the apogee. A trajectory and a view of the earth for an inclined orbit are presented in Fig. 2.

Not only the elevation variation but also the azimuth variation needs to be considered for coverage and antenna pointing purposes. For a unity range (assumed), the azimuth and elevation variation for the Molniya 12 hour orbit are shown in Fig. 3; note that the diagram is not to scale and it is included to exhibit antenna pointing only.

5 EFFECT OF TERRAIN

5.1 Selection of Site

To study the possible impact of terrain on coverage Seal Rock was chosen for analysis for the following reasons:

* It is representative of the West Coast of North America and Northern Europe
* The site has water, flat terrain, and mountainous terrain (Olympic Peninsular) in its vicinity
* Terrain data *(3 arc-second)* is available

The coordinates of Seal Rock, WA are 47°:43' N and 122°:53' W. A map of the area in the proximity of Seal Rock, WA is shown in Fig. 4, and a two-dimensional topographic plot with contours is shown in Fig. 5.

5.2 3-D View of Terrain Area

A 3-dimensional plot of the area (17x20 mi^2) shown in Fig. 5 presented in Fig. 6. The north-west view shows constant - elevation *above mean sea level (AMSL)* contours. (Note that the vertical scale is in *feet*, and the horizontal scales are in *miles*.) From topological plots of Figs. 5 and 6 it can be seen that there is water with a few islands to east of the site, but land and steeply rising mountains to the west. The highest mountains lie to the northwest of Seal Rock, WA some of these are over 6000 ft high.

5.3 Terrain Radials

For the purposes of propagation analysis it is more appropriate to examine the terrain data for the relevant azimuths. (Note 0° or 360° azimuth corresponds to North, 90° azimuth to East, etc.). Radial terrain data centered at Seal Rock, WA for 0°, 15°, 30°, ..., 345° azimuths every 0.1 mi was extracted from a 3 arc-second data base.

From Fig. 3, it can be seen that the azimuths 305°±15° and 50°±15° are relevant for the Molniya 12 hour orbit. Terrain elevation variations (ft. AMSL) along 300° and 315° azimuth radials are shown in Figs. 7 and 8 respectively. From these radials, assuming barren terrain (absence of foliage), the terrain elevation angle is ≈10° from Seal Rock, WA. Thus, if the satellite elevation angle is > 10° there is no effect of barren terrain on coverage duration.

5.4 Effect of Terrain on Coverage

The effect of barren terrain on coverage is shown in Fig. 9 at 30 GHz. A representative satellite elevation of 20° and terrestrial antenna elevation of 1 ft AMSL (e.g. for a submarine terminal) were assumed. Radial terrain data as described above was used. Each curve is an equi-field strength contour; the labels of the contour are relative values in dB's; the absolute value is meaningless. Bullington method was used to compute the obstruction loss. The path loss is determined by the degree to which obstructions penetrate the Fresnel zone. Effect of the mountains (Olympic Peninsular) to the north-west of Seal Rock on propagation and coverage is evident from Fig. 9.

Fig. 10 is similar to Fig. 9, except that the antenna height has been increased by 60 ft. A comparison of -120 and -114 dB signal contours in Figs. 9 and 10 respectively shows that signal level at given distance increased by approximately 6 dB with the increase in antenna height. As expected there is no relief in propagation loss in the north westerly direction.

The field computations in Figs. 9 and 10 do not include the effects of multipath or foliage. The signal probability distribution will depend on the earth terminal surroundings. Indeed it is conjectured that as the satellite begins to come into view and rises the signal statistics will change from log-normal shadowing to Rayleigh to Rician probability distributions.

To summarize, the distant irregular barren terrain even when it is steep has an elevation of 10° only and does not reduce the satellite angular (elevation) view. The irregular terrain however introduces an obstruction loss as shown in Figs. 9 and 10. More important are the terrain and foliage characteristics in the ***immediate*** vicinity of the terrestrial antenna; effects of these are considered in the following sections.

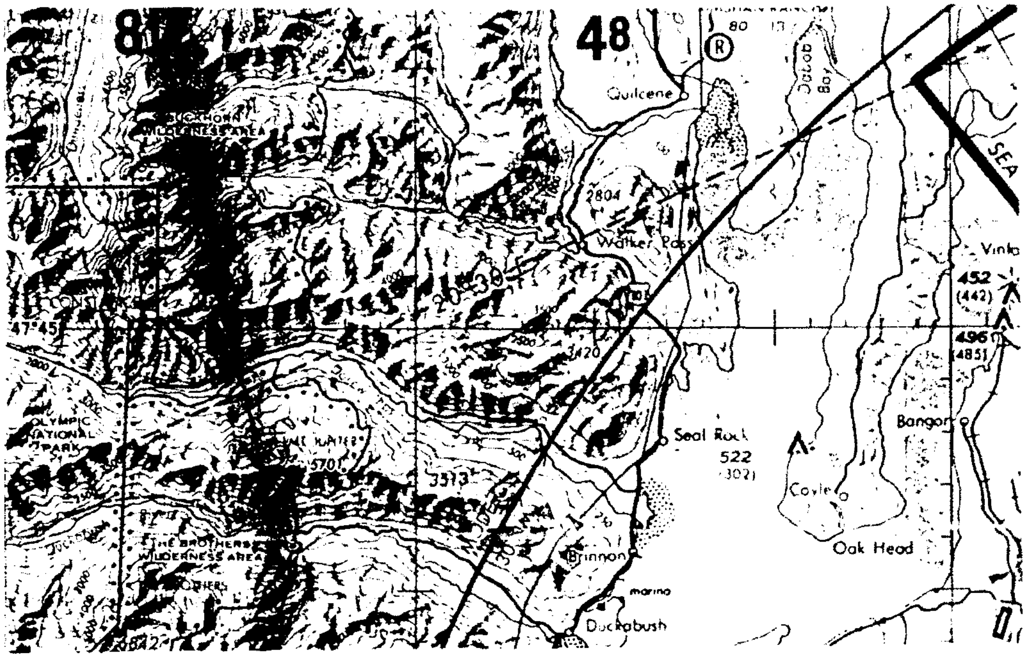

Fig. 4. Map of Seal Rock, WA.

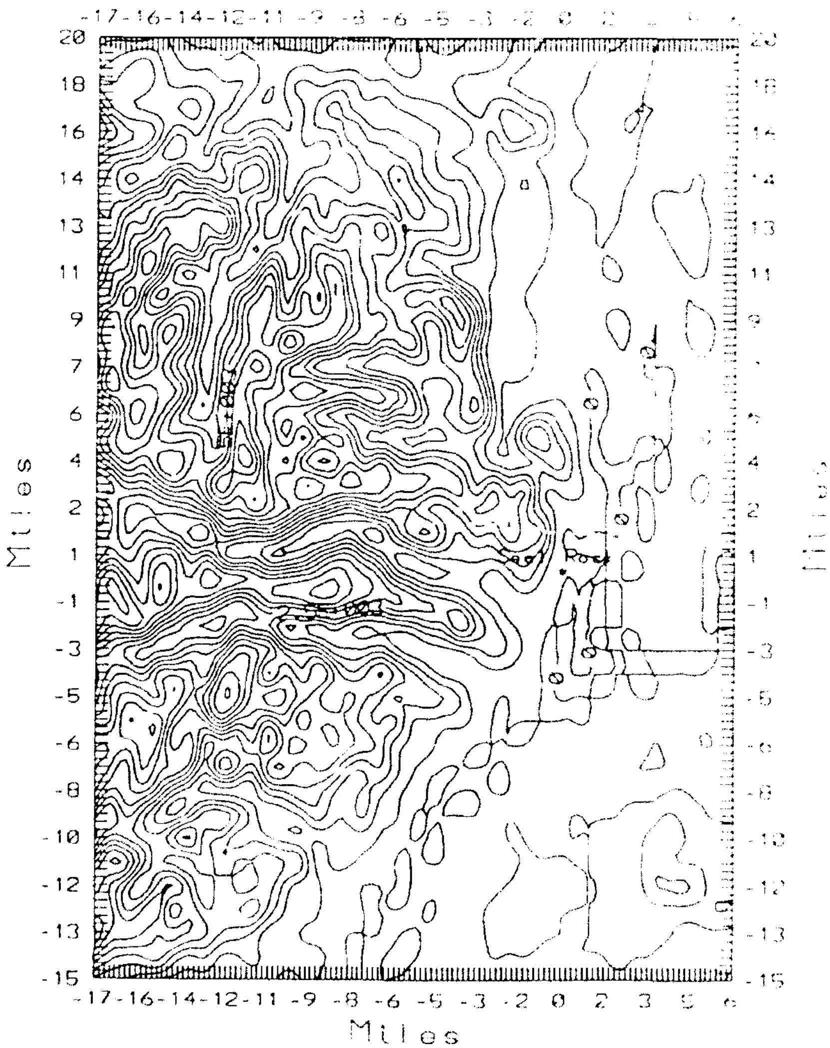

Fig. 5. Topographic contours of the Seal Rock, WA area.

Fig. 6.NW View of Seal Rock, Washington State, US

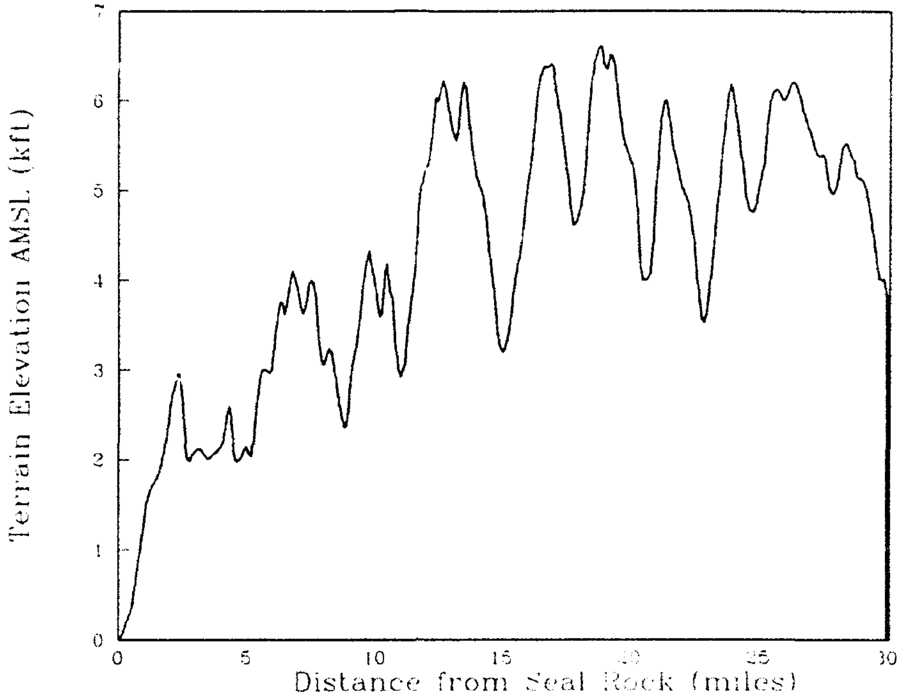

Fig. 7. Terrain elevation along [illegible] azimuth (radial).

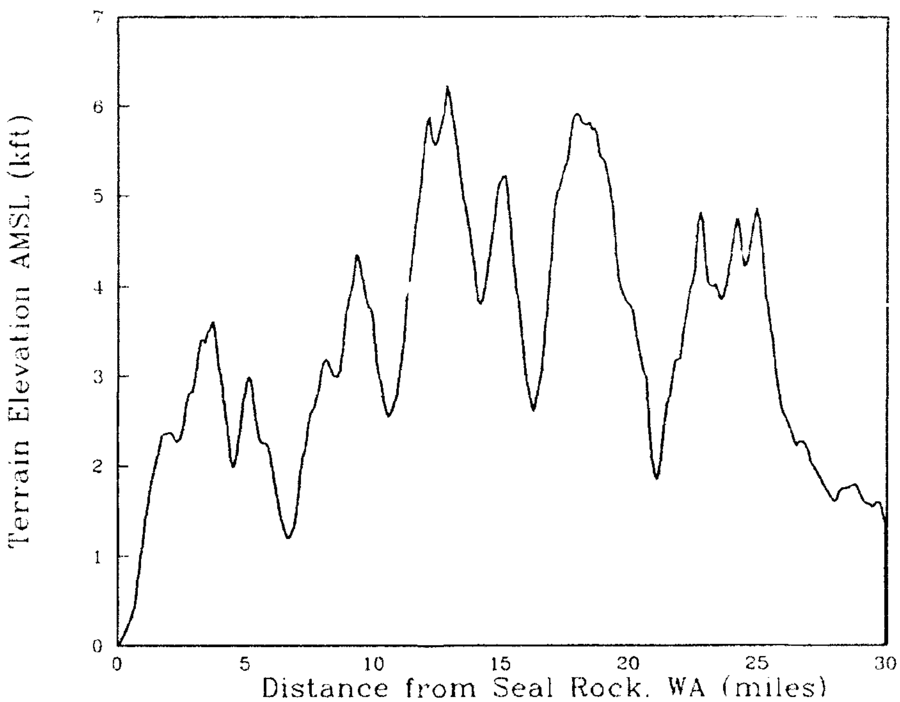

Fig. 8. Terrain elevation along 315 azimuth (radial).

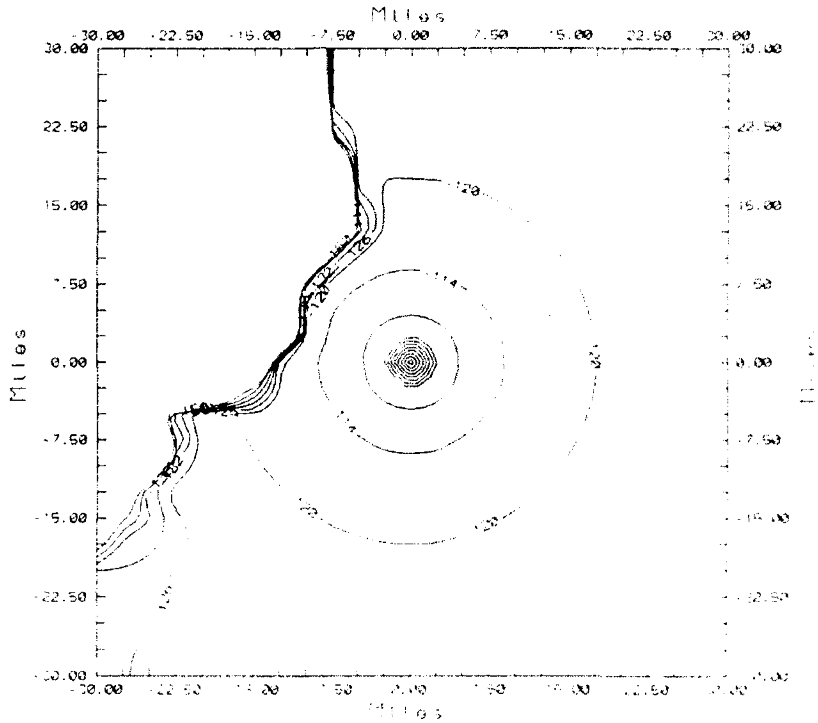

Fig. 9. Equi-field strength contours; antenna 1' AMSL (e.g. submarine).

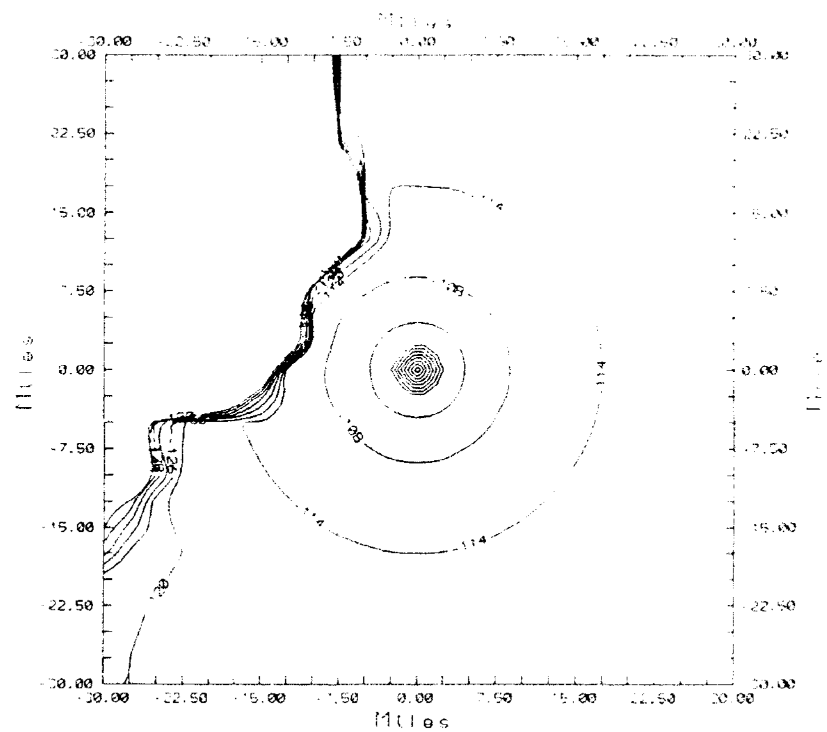

Fig. 10. Equi-field strength contours; antenna 61' AMSL.

6 EFFECT OF FOLIAGE

6.1 Foliage Penetration Loss

There is a paucity of data on propagation loss through foliage at millimeter waves. Some information is available[5, 7-10]. In general the propagation loss through foliage increases with frequency[7] reaching formidable value at 30 GHz of ≈5 dB/m. Attenuation due to foliage is given by[7]:

$$\alpha = k \quad f^{0.75} \quad \text{dB/m}$$

where, k is in the 0.25-0.50 range

f is the frequency in GHz

6.2 Effect of Foliage on Satellite View

The propagation loss due to foliage depends on the height of foliage and its distance from the terrestrial antenna as these factors determine the path length through the foliage, which in turn determines the minimum useable elevation angle of the satellite orbit. The foliage loss produces a loss (depression) in the useable elevation angle. A plot of *loss in useable elevation angle* vs. *foliage attenuation that can be tolerated* is shown in Fig. 11 with foliage height as parameter at 30 GHz.

From Fig. 11, it can be seen that even with a very generous margin of 10 dB for foliage loss only ≈2 m of propagation path length through foliage can be tolerated, furthermore a ≈2° depression in view angle below the tree tops is sufficient to consume the margin! That is, reliable propagation at ***mm waves*** can be achieved only when the satellite is visible above the tree tops.

6.3 Combined Effect of Foliage & Terrain

The combined effect of foliage and irregular terrain can be substantial; particularly if the foliage is tall and the terrain rises sharply in the vicinity of the terrestrial terminal. Near the west coast of Canada and USA the foliage consists of coniferous trees ≈100 ft tall, and the terrain is mountainous.

The elevation view angles of 60 ft high foliage at 30 ft, 45 ft, 60 ft, 75 ft and 90 ft distances from the terrestrial terminal are 63°, 53°, 45°, 39° and 34° respectively on a flat terrain. However, if the foliage is on terrain with 5° grade, the view angles are 68°, 58°, 50°, 44° and 39° at distances of 30 ft, 45 ft, 60 ft, 75 ft and 90 ft respectively.

The combined effect can be to limit the unobstructed satellite useable view to elevation angles of >45°! This suggests that satellite orbits that provide high elevation angle be used, the terminal be used only when the satellite elevation angle is high, and the terrestrial terminal be set up *far far* away from foliage i.e. in a clear area. Incidentally, the requirement to set up the terminal in a clear area could compromise covert operation under battle conditions.

6.4 Comments

To recapitulate:

* In barren environment the useable elevation angle is unaffected
* Distant mountainous terrain does not significantly reduce the useable duration of the satellite
* Presence of vegetation, trees specifically and particularly in the vicinity of the terrestrial terminal, reduce the useable duration of the satellite
* Tree covered mountainous terrain exacerbates the effect of trees by reducing the useable duration of the satellite
* Manpack, transportable and land-mobile terminals are affected by terrain and trees, but the airborne terminals and shipborne terminals at open sea are not affected
* In the presence of trees the fading might change from log-normal shadowing to Rayleigh to Rician as the moving satellite comes into view and rises
* Effect of foliage of different kinds and heights needs to be studied

7 SUMMARY & CONCLUSIONS

* Elevation angles for satellites in 5 different orbits at Seal Rock, WA have been examined. Expectedly the visibility of the satellites in the 5 orbits is not identical. CIS and GPS elevations angles are low and the satellites are available for shorter duration. Molniya 24 has high elevation and it is visible for a substantial part of the day.
* Terrain and coverage prediction analysis has been done by extracting terrain from a 3 arc-sec data base and using the Bullington method for computing the field. Effect of terrain on propagation loss as well as the effect of raising (/lowering) the antenna elevation has been shown.
* There is a paucity of propagation loss data in general and through foliage in particular at mm waves. Measurements need to be made to determine the probability distribution of the signals. Knowledge of the statistics of the signal will aid in establishing realistic fade margin that is neither onerous nor too low.

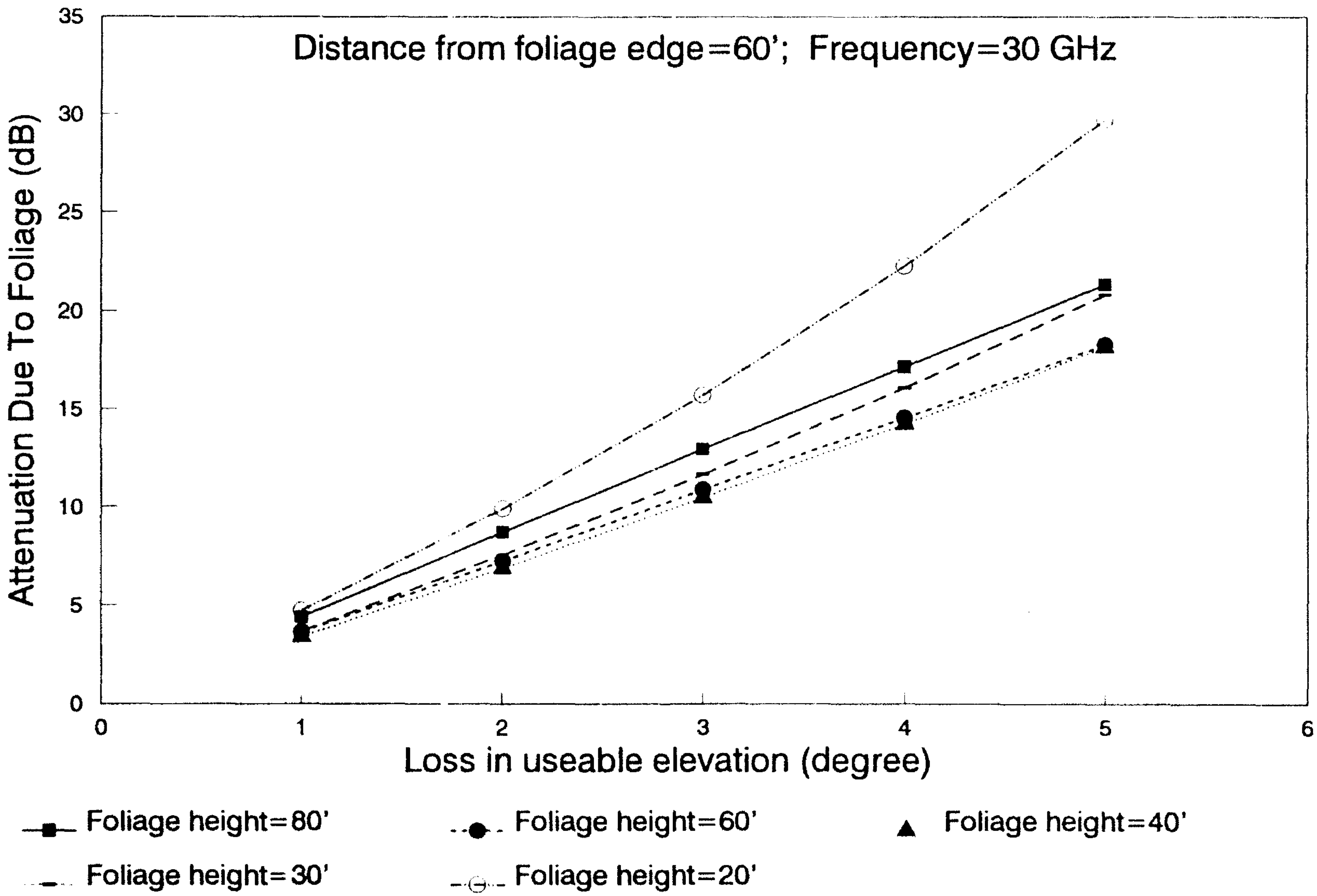

Fig. 11. Foliage attenuation vs. Loss in elevation; foliage ht. as parameter.

* Much work has been done to determine the effect of rain on EHF propagation, but the effect of terrain and foliage together has not been studied. There is a need to augment the traditional *"footprint"* estimation with propagation prediction using topographic and foliage data

7.1 Impact on Earth Terminals and Satellites

The earth terminals will have GPS or some other subsystem to determine the position, which can be communicated to the satellite. The incorporation of 3 arc-second terrain data base ***along with data on vegetation and man-made structures*** in the satellites will aid in determining a suitable time-window for communications. At the very least consideration of terrain and vegetation losses in link budget (EIRP, data rate, etc.) will make the system design more complete and thus improve reliability of TACSATS.

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LAUNCH VEHICLES FOR LIGHTWEIGHT TACSAT DEPLOYMENT

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1. SUMMARY

The need for tacsats is discussed and typical tacsat mission requirements are identified. A spectrum of potential tacsat launch systems is identified and characterized, including both US and non-US systems, and ranging from existing operational vehicles to very advanced vehicles that will not become available until well into the next century. The six basic launch system design drivers: performance, cost, operability, launch responsiveness, launch flexibility, and survivability are discussed as they relate to tacsat launch requirements. The launch systems that are described are categorized into: (a) mainline launch systems, (b) small fixed/relocatable/mobile launch systems, (c) ballistic missile-derived launch systems, and (d) far-term launch systems. The impact of the US/CIS Strategic Arms Reduction Talks (START) agreements on the availability of surplus ballistic missiles is discussed. The availability and utility of a wide selection of global launch sites are also examined. It is concluded that the general tacsat mission is a viable concept, and possibly satisfies a very critical future need. It is demonstrated that the tacsat launch services needs into the foreseeable future can be supported by US current and planned launch systems launched from existing US launch sites. However, numerous other options are available and should be investigated by the tacsat system designer/planner for viability and cost effectiveness.

2. INTRODUCTION

The North Atlantic Treaty Organization (NATO) alliance must respond to the change from a fairly stable geopolitical environment, with a well-defined adversary, to a very dynamic environment where the adversary and the theater of operations could change several times during the lifetime of some of the current tactical satellites and where conflict could arise in more than one theater at once. The evolving world situation also suggests that there will be a need for less emphasis on new strategic systems and more emphasis on new tactical systems. It has been proposed that what have been called "lightsats" or "cheapsats," but what are termed "tacsats" in this paper, could be candidates for satisfying new evolving tactical mission requirements. Such satellites would place new requirements on space launch services, which historically have evolved in response to requirements for larger and larger payloads on orbit, whether their mission is tactical or strategic. Use of the current US mainline launch vehicles, for instance, would entail launching tacsats either as multiple payloads, or as auxiliary (piggyback) payloads on a flight whose primary payload is a conventional large satellite. In current practice in the US, the latter option would not generally be available on Department of Defense (DOD) flights since DOD payloads are normally designed to utilize the full payload capability of the launch vehicle. Conventional scheduled flights would have to be re-planned to fly off-loaded in order to permit tacsat payloads to be added if a sudden crisis generated an unscheduled need. The establishment of such a policy could impact the payload requirements for upgraded Medium Launch Vehicles (MLVs) or the new National Launch System (NLS). However, planning to launch tacsats as single payloads on small launchers (fixed, relocatable, or mobile) is not necessarily the most cost-effective option.

Instead of separately designing and developing the orbiting vehicle system, which is later integrated with a separately designed and developed space launch system, the tacsat option opens up for consideration the concurrent design and development of a single system of which the orbiting vehicle and the launch vehicle are merely modular elements. The concept of using the launch vehicle upper stage as a spacecraft bus is not new and was used successfully by Lockheed in their Agena program. There is some evidence for believing that such weapon systems as Nike and the air-cruise missiles are inherently more operable than conventional space launch systems. If the design philosophy described above (to consider the orbiting subsystem and the launch subsystem as elements of a single, integrated system) is adopted, it may be possible to improve the operability and lower the cost of satisfying the same tactical mission requirements by the historically conventional parallel system development approach. It is conjectured, for instance, that a variety of missions could be satisfied by selecting and assembling different systems from a limited inventory of standardized (modular) orbiting vehicle and launch vehicle subsystem elements.

Also included as candidate launchers are the strategic ballistic missiles that will be retired as a result of the emerging START treaty agreements. Special limitations on the conversion of ballistic missiles to space launchers apply and need to be considered.

It is evident that the tacsat concept introduces a number of complex new issues that need to be addressed and that must be placed in the context of a total cost-effective system solution to a dynamically evolving threat. However, this paper does not propose to resolve these broader issues but is intended to describe the launch vehicle options for launching lightweight tactical satellites to satisfy a variety of potential missions. The potential missions and the general characteristics of tacsats that might satisfy these missions are delineated. The important factors in the identification of candidate launch systems for these missions, i.e., weight, dimensions, orbits and constellation architecture of the satellites; and performance, cost, operability, launch responsiveness, launch flexibility, and survivability of the launch system, are characterized. Requirements for launch responsiveness and launch flexibility are not well defined, but a spectrum of options is discussed.

The launch systems are grouped into (a) mainline launch systems; (b) small fixed, relocatable, and mobile systems, (c) ballistic missile-derived systems, and (d) far-term systems. The primary focus is on US launch vehicles and launch from the continental US, but the capabilities of other nations' launch services and launch sites are also described. This is because it is unlikely that the US, or even NATO, will take military action that would involve unilateral full-scale use of the tacsat concept as discussed herein. Although resisting local despots needs single-minded leadership, and only individual governments can truly provide that, the US and NATO must coordinate offensive action with international organizations such as the European Community (EC), the United Nations (UN), the Western European Union (WEU), the Conference on Security and Cooperation in Europe (CSCE), and with other nations, including the several ex-Soviet Bloc nations that are indicating an interest in joining NATO or the EC for mutual military and economic security. The global market for, and utility of, tacsats must also be examined in the light of the "Open Skies" treaty signed on the 25 March 1992, in which 25 countries, including all the 16 NATO countries, 5 East European ex-members of the Warsaw Pact, and Russia, Ukraine, Belorussia, and Georgia agreed to permit airborne overflights of their territory.

3. LAUNCH SYSTEM SELECTION CRITERIA

Tacsats can either be launched on demand, launched on schedule as primary payloads, or launched as payloads of opportunity (auxiliary payloads flown piggyback on regularly scheduled flights). The tacsat mission raises the issue of satisfying launch system design requirements that may possibly have not been satisfied or emphasized in the past. Traditional launch and missile system design drivers can be roughly divided into six categories: (a) performance, (b) cost, (c) operability, (d) launch responsiveness, (e) launch flexibility, and (f) survivability. The members of the three major families of mainline US expendable launch vehicles; Atlas, Delta, and Titan, are examples of performance-driven designs. The US Space Shuttle was planned to be a cost-driven system but, in common with all launch systems developed up to the present, did not achieve its goal of low-cost access to space. The US Advanced Launch System (ALS), now replaced by the NLS, was conceived as being both an operability-driven and a cost-driven system. The solid propellant ballistic missiles, such as the US Minuteman III (MM-III), are examples of launch responsiveness-driven systems and therefore any space launch derivatives of ballistic missiles will inherit this design characteristic. The air-mobile systems, such as the US Pegasus and the proposed Commonwealth of Independent States (CIS) Space Clipper, represent systems that provide the launch flexibility to select a beneficial location for a specific mission and an all-azimuth launch capability.

Survivability is an additional design driver that has historically been of importance to weapon delivery systems. At present, survivability does not appear to be important to tacsat launch systems, but this factor could become more important in the future as more developing nations gain access to sophisticated offensive weapons.

4. REQUIREMENTS

4.1 Missions

The overall tacsat mission is characterized by the need to augment capital space assets during times of crisis; the need to surge at the outbreak of a major conflict; and the need to reconstitute space systems that have been damaged or destroyed. Tacsats satisfy a military need to provide space systems that are dedicated to support tactical commanders without inflicting a heavy logistic and administrative responsibility. Ideally, the battle commanders need the operational control to task satellites as their own assets to ensure direct and timely access to real-time data and information from space. Use of the space systems should be an intrinsic part of routine peacetime military training exercises and not brought into play only to respond to an actual war-fighting emergency.

Tacsat weights can range from less than 50 lb (23 kg), in the case of the Defense Sciences International Microsat, to the 150 lb (68 kg) Macsats which were used with success by US Marine units during the Persian Gulf War, to 1,500 lb (680 kg) Iridium-like satellites. One arbitrary definition of a tacsat is that it should weigh less than 2,200 lb (1,000 kg) and be launched for less than \$20M, (although some commercial lightsats have already been launched for well under \$1M). A better goal might be a weight of less than 1,500 lb (680 kg) and a launch cost of \$15M.

Table 1 shows a potential set of tacsat launch requirements. The missions include a broad range of capabilities that a field commander needs, including communications, weather, land and ocean surveillance, theater surveillance, and missile warning. In most cases, the tacsats would serve to augment the larger conventional satellites by providing the field commanders with dedicated assets, and may be used to reconstitute unexpectedly failed satellites on a temporary basis. Treaty monitoring is included, although this may be more a support function to, for instance, the United Nations Security Council than the battlefield commander.

Table 1 Potential Tacsat Launch Requirements

MISSIONS	SURGE INTERVAL	ORBITS	WEIGHTS [illegible]	REQUIRED LAUNCH RESPONSIVENESS
COMMUNICATIONS • SURGE [illegible] • AUGMENT/BACKUP [illegible] SYSTEMS	[illegible]	[illegible] ELLIPTIC [illegible]	[illegible]	NONE – STORE ON ORBIT PRIOR TO HOSTILITIES
ENVIRONMENTAL SENSING • AUGMENT NEAR CONTINUOUS WEATHER SATS	[illegible]	GEOSYNCH SUN-SYNCHRONOUS	454 (1000)	NONE – STORE ON ORBIT
• RECONSTITUTE UNEXPECTED SATELLITE FAILURES		WALKER [illegible] POLAR	[illegible]	NOT ESTABLISHED REQUIREMENT ([illegible] days)
THEATER SURVEILLANCE • ENEMY LOCATION TYPE, SIZE • STATUS OF AIRFIELDS RAILROADS, BRIDGES	[illegible]	WALKER ALL INCLINATIONS INCLUDING SUN SYNCHRONOUS	[illegible]-454 (500-1000)	NOT ESTABLISHED REQUIREMENT ([illegible] days)
MISSILE WARNING • DETECT, TRACK TBMs • AUGMENT DSP	[illegible]	GEO OR HIGHLY ELLIPTIC [illegible]	[illegible]	NONE STORE ON ORBIT
TREATY MONITORING • AUGMENT CURRENT SYSTEMS WITH RAPID NO-NOTICE INSPECTIONS		WALKER ALL INCLINATIONS OR SUN SYNCHRONOUS	454 [illegible]	NOT ESTABLISHED REQUIREMENT ([illegible] hours)
[illegible] AND SENSING • TIMELY MULTISPECTRAL EARTH IMAGERY	[illegible] DAYS	WALKER (300-500 nmi) SUN SYNCHRONOUS ALL INCLINATIONS	454 (1000)	NOT ESTABLISHED REQUIREMENT

(550-925 km (300-500 nmi)

It is seen from Table 1 that the orbits for the various missions fall into two general categories: (a) geosynchronous (GEO) or highly elliptic for continuous coverage missions; and (b) low earth orbit (LEO) sun-synchronous for all inclinations (to cover any potential target area) for Earth surveillance missions. High altitude satellites would probably be stored on orbit prior to hostilities. An attempt is made to quantify launch responsiveness requirements. Satellite weights are based on very preliminary tacsat designs, and should in no way be considered firm requirements.

4.2 Constellations

Tacsat constellations can range from single satellites to complex multiple satellite global constellations, such as an Iridium-like global communications constellation that includes 77 satellites in 7 orbital planes. Since there could be scenarios where multiple tacsats need to be launched at one time, potential launch options cannot necessarily be confined to small launch systems. Opportunities for payload manifesting must be examined in order to improve cost effectiveness. It is conceivable, for instance, for a whole plane of 11 Iridium-like tacsats to be deployed on a single mainline launch vehicle.

4.3 Payload Manifesting

For constellations with a large number of small satellites in a large number of orbital planes, manifesting on mainline launch vehicles is an attractive option. At low-to-mid inclinations, the most effective means of distribution is to use orbital regression. Spacecraft are launched into a park orbit separated in altitude from the mission orbit. Earth oblateness causes a natural drift, or regression, of the orbits, and the rate of drift is a function of the altitude (and inclination) of the orbit. Thus the differential regression rates between the two orbits permit a phasing of the orbits to take place over time. At the appropriate time, with planes aligned, a spacecraft destined for a certain ring performs an inter-orbit transfer into its respective mission orbit. Regression rates, however, are reduced at high inclination and, in fact, are zero for polar orbits. Thus, for high inclination constellations, this approach is not practical.

A second strategy is to launch the spacecraft into the same altitude as the mission orbit, but at a lower inclination, thus establishing a relative regression rate based on the differential inclination. The plane-change maneuver to inject the spacecraft into the mission orbit is a high energy maneuver, however, and the energy required increases rapidly as the magnitude of the plane change increases. The plane-change maneuver implies the use of propellant, which reduces the number of satellites that can be manifested on a single launch. A balance between plane change and acceptable regression time must be established to make effective use of the launch system performance.

4.4 Launch Responsiveness

A further consideration is the launch system responsiveness. For tacsat missions, there is an implication of high responsivity of the satellite system, both on the ground and once it is in orbit. In addition to launching the first satellite rapidly, there is the issue of launch repetition rate requirements (the need to reload the system). This may be solved with additional storage, test and launch pad facilities, or rapid recycling of the pad, or most probably a combination of both. In every case, the more responsivity required, the higher the non-recurring cost for facilities and equipment and the higher recurring cost for manpower and maintenance. Trade studies need to be made when launch responsiveness requirements are better defined.

Ballistic missile-derivatives offer the possibility of launching in a matter of seconds if they are maintained on alert status with the satellite in the same state of readiness. This does not seem to be a necessary requirement for tactical situations, at least in the near-term, since there is a low probability of the rapid counterforce attack assumed in the case of strategic ballistic missiles. A more reasonable approach would be to store the vehicles under conditions that would allow them to be brought into a state of launch readiness in a matter of days; they could be launched on demand in a matter of hours if desired.

From the ultimate responsivity of the ballistic missile-derivatives, the next most responsive systems are the small, fixed, relocatable, or mobile launch systems. It is anticipated that these systems could be launched in a matter of days without a huge investment. Finally, the current medium and large launch vehicles have long callup times because they are not designed for high responsivity or recycle times. Therefore it would take a large investment in launch site facilities to bring their responsivity into the one to two week range.

5. MAINLINE LAUNCH SYSTEMS

There are presently four nation-states that offer routine global mainline space launch services: the United States, Europe, the Commonwealth of Independent States, and the Peoples Republic of China. Technically, Japan also has a mainline launch system, the H-1. However, this vehicle is currently being phased out in favor of the H-2 and made its last flight in February, 1992. When the H-2 becomes operational, five nation-states will be able to provide mainline launch services. However, difficulties being experienced with the LE-7 cryogenic first-stage engine could delay the predicted 1993 initial launch capability date.

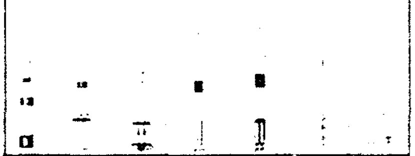
5.1 Current and Near-Term

5.1.1 *United States (US)*

The current US mainline launch system families include the Atlas, Delta, and Titan families. The Atlas, Delta, and Titan II are medium launch vehicles, with payload capabilities of between 6,000 and 20,000 lb (2,728 and 9,091 kg) to LEO. Heavier lift capability is provided by the Commercial Titan III and the Titan IV, which can lift close to 48,000 lb (21,818 kg) to LEO.

This group of launch vehicles is the most viable for the high energy tacsat missions, not only because of payload capability, but because the suggested store-on-orbit strategy for the GEO satellites is consistent with their current low response times. The Titan II can currently be flown only from the west coast launch site at Vandenberg Air Force Base (VAFB) in California. The Delta II can deliver 2,100 lb (910 kg) and the Atlas II, 3100 lb (1,410 kg) to GEO. For a 1,000 lb (455 kg) satellite, this translates into about $25 M per payload launched two at a time on Delta or three at a time on Atlas, while for 1,500 lb (682 kg) satellites, these numbers become about $50 M for a single satellite on Delta and $40 M each for two satellites on Atlas. The viability of launching multiple satellites to GEO would have to be assessed by the users, but it would appear to be a reasonable approach since there are three GEO missions identified, and, in any case, silent spares can be deployed if sufficient excess booster capability exists. The benefit of controlling satellite weight, without compromising mission capability, is apparent.

Figure 1 summarizes the payload capabilities of the current US medium and large mainline expendable launch vehicles to various orbits of interest and several other characteristics, such as payload accommodation, reliability, and cost per flight. Launch rate capacity circa 1995 is also shown.



LAUNCH VEHICLE	ATLAS E	TITAN II	DELTA II 7920/7925	ATLAS II	ATLAS IIAS	COMM'L TITAN III	TITAN IV SRMU
RESPONSIBLE AGENCY	USAF	USAF	USAF	USAF	COMM'L	COMM'L	USAF
PRIME CONTRACTOR	GD	MMC	MDC	GD	GD	[illegible]	MMC
PERFORMANCE (lb)							
LEO POLAR	1,800	4,200	7,470	11,900	15,000		41,000
LEO DUE EAST			11,100	14,100	8,500	32,000	47,100
GTO			4,010	5,900	7,700	9,500[4]	[illegible][4]
GEO			2,100	3,100	4,100	3,000[4]	[illegible][4]
RELIABILITY %	92.0	91.5	[illegible]	[illegible]	NOT FLOWN	[illegible]	[illegible]
PAYLOAD ACCOMMODATION							
PAYLOAD DIA ft	7.0	9.3	9.2	9.5, 12.0	9.5, 12.0	12.0	15.0
CYLINDER LENGTH ft	4.9	12.0, 17.0, 22.0	12.0/8.3	17.8, 13.1	17.8, 13.1	29.4	40.0
CONE LENGTH ft	7.8	[illegible]	[illegible]	12.5, 19.2	12.5, 18.2	[illegible]	[illegible]
INITIAL LAUNCH CAPABILITY	1974	1988	1990	1991	1993	1990	1990
LAUNCH SITE	ELS	WLS	ELS/WLS	ELS	ELS	ELS	ELS/WLS
LAUNCH CAPABILITY (1995)	N/A	3	128	9	9	4	62
RESPONSE TIME	HIGH	HIGH	HIGH	HIGH	HIGH	HIGH	V. HIGH
COST/FLT ($91 M)	48	43	41-52	74-84	116-128	130-150	154-227

1) 28.5
2) With TRANSTAGE
3) With KICK MOTOR
4) CENTAUR UPPERSTAGE (other options: IUS, [illegible])

Figure 1 Current US Mainline Launch Systems

5.1.1.1 *Delta family of vehicles*

The Delta family of vehicles was derived from the Thor Intermediate Range Ballistic Missile (IRBM) by adding several small Solid Rocket Motors (SRMs) and the Delta second stage. Deltas are used to launch payloads into GEO and LEO from Eastern Test Range (ETR) launch complexes LC-17A and LC-17B, and to polar orbit from the Western Test Range (WTR) space launch complex SLC-2. The capabilities have grown through a series of upgrades. The prime contractor for the Delta family of vehicles is McDonnell Douglas, Huntington Beach, California.

The Delta II was selected as the Air Force Medium Launch Vehicle I (MLV-I) and is currently used for launching the

NAVSTAR Global Positioning Satellite (GPS), Space Defense Initiative Organization (SDIO) experimental payloads, NASA scientific payloads, and commercial payloads. The initial version (Delta 6925) first flew in February, 1989, followed by an upgraded version (Delta 7925) in November, 1990. Delta 7920 has nine Graphite Epoxy Motor (GEM) strap-ons and supports the heavier weight of current GPS satellites. A two-stage version exists for each Delta II: Delta 6920, which is capable of placing 8,800 lb (4,000 kg) into LEO due east; and Delta 7920, which is capable of placing 11,100 lb (5,045 kg) into LEO due east. The main difference between the Delta II vehicles (6920 versus 7920) lies in the use of Castor Solid Rocket Motor (SRM) strap-ons (steel cases) versus the use of GEM strap-ons (graphite epoxy cases). The Delta 7925 provides performance to transfer orbits and uses a third stage. This third or upper stage is a modified PAM-D, with a Star-48B solid motor mounted on a spin table to spin-up the stage/payload combination before deployment. Incorporation of this third stage provides the capability to deploy 4,010 lb (1,820 kg) of payload into GTO (without the third stage, only 2,800 lb (1,270 kg) can be deployed into GTO). An additional spacecraft kick motor is required for geosynchronous orbits.

5.1.1.2 *Atlas family of vehicles*

The Atlas is a former Air Force Intercontinental Ballistic Missile (ICBM) weapon system converted for use as a space launch vehicle. The Atlas D was man-rated and flew four successful missions during the Mercury program, including the first U.S. manned orbital flight by John Glenn. The system concept, now over 30 years old, has gone through a series of modernizations and upgrades, which have enhanced its payload performance capability considerably. There are several Atlas versions, existing and planned, but common to all is the use of one and a half liquid propellant stages. Both stages ("booster" and "sustainer") are ignited on the ground and burn in parallel. After the booster engines are jettisoned, the sustainer engine continues burning to orbit. The prime contractor for the Atlas family of vehicles is General Dynamics Corporation, San Diego, California.

Atlas E. The Atlas E is a DOD launch vehicle presently used to launch smaller payloads to low polar orbit from space launch complex SLC-3 at Vandenberg Air Force Base (VAFB). The vehicles are former Atlas ICBMs that have been decommissioned, refurbished, modified, tested, and certified for space flight. The Atlas E is primarily used to support the DOD Defense Meteorological Satellite Program (DMSP) and the National Oceanic and Atmospheric Agency (NOAA) satellites. It can place 1800 lb (880 kg) into low polar orbit. Only a few vehicles remain in inventory.

Atlas II. The Atlas II is used for communication satellite launches such as the Defense Satellite Communications System (DSCS-III). It was selected as the Air Force Medium Launch Vehicle II (MLV-II) and is designed to perform LEO and GTO missions. It consists of the booster and a Centaur upper stage and can place 14,100 lb (6,410 kg) into LEO due east, 5,900 lb (2,680 kg) into GTO, or 3,100 lb (1410 kg) into GEO using a kick stage.

The Atlas contractor is offering four versions of the Atlas: Atlas I, Atlas II, Atlas IIA, and Atlas IIAS for launching NASA and commercial payloads. The Atlas IIAS is an improved version of the Atlas II that has solid strap-ons and in which the Pratt and Whitney RL-10 engine in the Centaur upper stage is increased in thrust from 16,500 lb to 20,800 lb (7,500 kg to 9,450 kg). The Atlas IIAS payload to GTO will be 7,700 lb (3,490 kg).

General Dynamics Commercial Launch Services has proposed carrying so-called "companion" satellites ranging from 990 lb to 2,970 lb (450 kg to 1350 kg) to LEO at 28.5 deg on launches of primary payloads.

5.1.1.3 *Titan family of vehicles*

The Titan family of vehicles is a series of Air Force vehicles that has evolved from the Titan ICBM system over the past 35 years. Three main launch vehicle configurations presently exist: Titan II Space Launch Vehicle (SLV), Titan III Commercial, and Titan IV, with variations of each. All have two core stages using liquid propellants. The last two use segmented SRMs for the initial stage to enhance performance. Composite-case SRMs (designated SRMU) will soon replace the steel-cased solids to provide increased performance. Additional proposals for increased performance include the use of liquid cryogenic propellant boosters, more and longer SRMs, and more liquid engines with stretched and/or larger diameter core stage tankage. The prime contractor for the Titan family of vehicles is Martin Marietta Corporation Astronautics Group, Denver, Colorado.

Titan II. The Titan II space launch vehicle was conceived by DOD to utilize an existing resource and, at the same time, augment the dwindling Atlas E launch vehicle inventory for launching smaller payloads to polar orbit. The Titan II vehicles (like the Atlas E vehicles) are former ICBM weapon systems that have been decommissioned, removed from their silos, refurbished, modified, tested, and certified for space flight. There were originally 55 Titan ICBMs in inventory, and the Air Force has a continuing program to modify and launch these vehicles as required. The initial launch capability of the Titan II was achieved in September, 1988. The Titan II can place up to 4,200 lb (1,910 kg) in LEO polar orbit from space launch complex SLC-4 (West) at VAFB. Proposals for upgrading Titan II performance include developing a Cape Canaveral capability, long duration circularization burns with added propellants, and SRM strap-ons. The addition of eight Castor IVs, for example, would launch 8,900 lb (4,045 kg) to LEO polar. Configuration studies include the addition of up to 10 GEM solid rocket motor strap-ons.

Titan II, in its ballistic missile configuration, can be launched with less than one minute of warning since propellant can be left for long periods of time without deterioration of the fuel tanks or the propulsion system.

Commercial Titan III. The Commercial Titan III is derived from the Titan 34D with a stretched second stage and a hammerhead (larger diameter) shroud for dual or dedicated payloads. The first commercial Titan III was launched in December, 1989, and can launch 32,000 lb (14,540 kg) into LEO. It is compatible with the McDonnell Douglas PAM-DII, the Martin Marietta Transtage, and the Orbital Sciences Corporation Transfer Orbit Stage (TOS) upper stages to provide GTO capability of 4080 lb (1,850 kg), 9500 lb (4,320 kg), and 11,000 lb (5,000 kg), respectively. Performance to GEO is approximately 5,500 lb (2,500 kg) using a spacecraft kick motor. Martin Marietta has examined a number of payload deployment schemes for launching small payloads, both as auxiliary payloads and as multiple primary payloads, but these design options have not been exercised.

Titan IV. The DOD's Titan IV development was begun as a complement to the Space Shuttle, but following the Challenger accident, the program was expanded to accommodate critical DOD payloads. The program was further expanded in 1987 as the full impact of the Shuttle delays and cancellation of the Shuttle-Centaur program became clear.

The Titan IV uses either a 7-segment SRM or a 3-segment SRMU. It is now capable of delivering 10,000 lb (4,550 kg) to GEO with the current 7-segment SRMs. When the SMRU boosters now under development are available, the Titan IV/Centaur will be able to launch 12,700 lb (6,350 kg) to GEO. The Titan IV/IUS is currently operational and is capable of delivering 5,250 lb (2,390 kg) to GEO. A Titan IV with no upper stage is used for launching into polar and high inclination orbits. This configuration can deliver 39,000 lb (17,700 kg) to LEO due east, or 31,000 lb (14,100 kg) to a 100 nm (185 km) polar orbit. The Titan IV could, in theory, fly auxiliary payloads but this is not a very practical option since its primary payloads normally use full payload capability.

5.1.2 *Europe*

Europe is represented by the Ariane 4 family, which provides a range of payloads from 8,000 to 21,000 lb (3,6360 to 9,545 kg) to LEO. There are six versions of Ariane 4, with a mix of liquid and solid propellant strap-on boosters. There are two subsatellite services, the SPELDA (Structure Porteuse Externe pour Lancement Double Ariane), Dedicated Satellite Service (SDS), and the Ariane Structure for Auxiliary Payloads (ASAP), plus different versions of the SPELDA payload fairing. In this way, it caters to a wide range of payloads. The vehicles are launched from a launch site at Kourou, French Guiana; a launch azimuth range of 104 deg is possible from 349.5 to 93.5 deg. The cost per flight for a single GTO launch is between $65M and $95M.

Fifty satellites have been launched using the SYLDA (Systeme de Lancement Double Ariane) or SPELDA dual launch capability, which has resulted in a satellite deployment rate of about 12 per year, up from 7 to 8 launches per year previously. The launch limit for the Ariane 4 is set at 10 flights per year, which allows for delays or problems. Three commercial microsatellites have been flown using the ASAP, and as many as six ASAP attachment points can accommodate multiple satellites up to about 91 lb (200 kg) each. The entire ASAP structure can be reserved for between $700,000 and $800,000. In order to satisfy customers' demands, a shortened version of SPELDA, named SDS (Spelda Dedicated Satellite) is being built by British Aerospace to accommodate a satellite in the 2,700 lb (6,000 kg) class, plus an additional satellite of up to 800 kg. The capabilities of the Ariane 4 are summarized in Figure 2.

Arianespace, the commercial consortium that operates Ariane, predicts the minisatellite market (commercial and military) to reach 20 per year by 1993, and its goal is to capture 50% of the market.

5.1.3 *Commonwealth of Independent States (CIS)*

The availability of ex-Soviet Union launch systems is complicated by the political changes that are taking place and the fact that space assets are no longer controlled by a single authority. At least five space agencies (three in Russia) appear to be emerging and at least three members of the Commonwealth (Russia, Kazakhstan, and Ukrain) are laying claim to the potential benefits of marketing space technology to the world. The distribution of the major CIS space assets are shown in Figure 3.

Russia appears to have control of most of the ex-Soviet arsenal of launch systems, including the Energia/Buran, the Kosmos, the Proton, the Tsyklon, and the Vostok/Soyuz/Molniya. However, the Zenit, which is the only ex-Soviet vehicle built specifically for commercial launch and may possibly be launched from the proposed Cape York Space Port in Queensland, Australia, is built by Yuzhnoye NPO in the Ukraine. The Yuzhnoye Design Bureau is also proposing the air-launched Space Clipper, which is discussed later. The launch capabilities of the ex-Soviet unmanned launch systems are summarized in Figure 4.

5.1.4 *Peoples Republic of China (PRC)*

The PRC is represented by the Long March family of vehicles. China's space launch services depends on a number of interacting organizations, each of which is responsible for part of the launch services package. The China Great Wall Industry Corporation (CGWIC) is responsible, under the Ministry of Astronautics, for coordination between foreign customers and the other elements of the launch services organization. The launch capabilities of the Long March family are summarized in Figure 5.

5.2 In Development and Planned

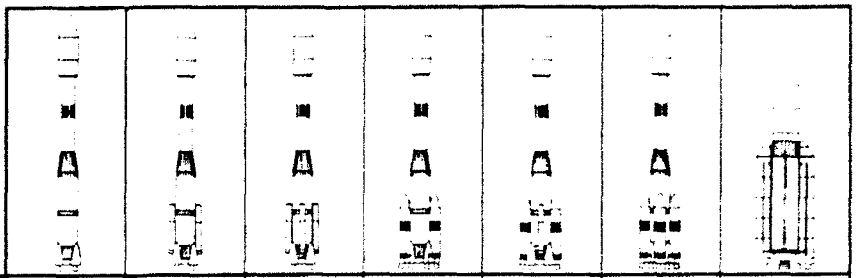
Three nation-states, the United States, Europe, and Japan have clearly laid out plans to extend their mainline expendable launch vehicle fleet.

5.2.1 *United States*

Medium Launch Vehicle 3 (MLV-3). The US Air Force plans to award a contract during fiscal year 1993 for the MLV-3, which would become available in 1996 and has, as its primary mission, the launch of GPS Block IIR. The MLV-3 is intended to "bridge-the-gap" until the NLS is available, and any performance improvements will be financed by private industry. This procurement could conflict with the procurement of the NLS, and a decision may have to be made in Congress to select one or the other. Although no characteristics of the MLV-III are releasable at this time, the procurement could provide an opportunity to plan for discretionary auxiliary payload deployment.

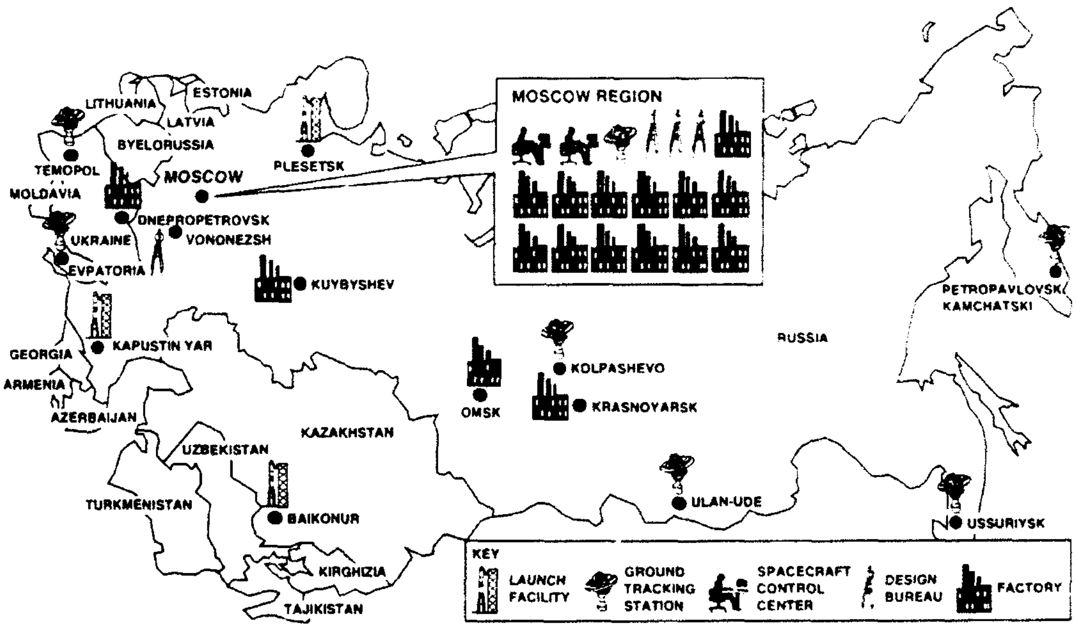
National Launch System (NLS). The NLS is using an evolutionary approach for the development of a family of launch vehicles and operational infrastructure with the capability to place a wide range of payloads into orbit at a fraction of current costs. The operability goals of NLS are to make space launch activities as routine as those of a long-haul trucking company.

The NLS vehicles currently serving as points of reference range from a small, two-stage vehicle capable of placing approximately 20,000 lb (9,090 kg) into LEO (the immediate focus), to a one and one-half stage vehicle utilizing a Shuttle External Tank (ET) derived tank section (common core) for moderate sized payloads to LEO, and capable of being upgraded to a two and one-half stage vehicle (by adding an upper stage) with GEO capability; to a heavy lift configuration using ASRMs as strap-on boosters, the common core, and a cargo transfer vehicle (CTV) for cargo delivery to Space Station Freedom (SSF). A new NLS upper stage, used with the one and one-half stage vehicle for GEO missions, is also being considered for use as the final stage of the 20,000 lb (9,090 kg) (NLS-3) vehicle to further provide commonality. Vehicles with increased payload capability, attained via modular growth to meet heavy lift requirements up to 124,000 lb (56,360 kg), are also under study. Payload estimates for the various members of the NLS family meet both NASA and DOD requirements. The primary interest of the DOD in the one and one-half stage is to place 50,000 lb (22,730 kg) into an 80 x 150 nm (148 x 278 km) orbit; NASA's interest is to use it to deliver 55,000 lb (25,000 kg) of net payload to the SSF. The DOD plans to use the two-stage vehicle to deliver 20,000 lb (9091 kg) to an 80 x 150 nm (148 x 278 km) orbit, 4000 lb (1,818 kg) to GEO, or 8,000 lb (3,636 kg) to GTO; NASA would use it to deliver 18,000 lb (8,182 kg) of net payload to the SSF. The two and one-half stage vehicle would deliver 97,000 lb (44,090 kg) to the 80 x 150 nm (148 x 278 km) orbit or 15,000 lb (6,818 kg) to GEO; it could also deliver 83,000 lb (37,730 kg) of gross payload to the SSF. The HLLV option could deliver 135,000 lb (61,360 kg) to the 80 x 150 nm (148 x 278 km) orbit or 124,000 lb (56,364 kg) of gross payload to the SSF.



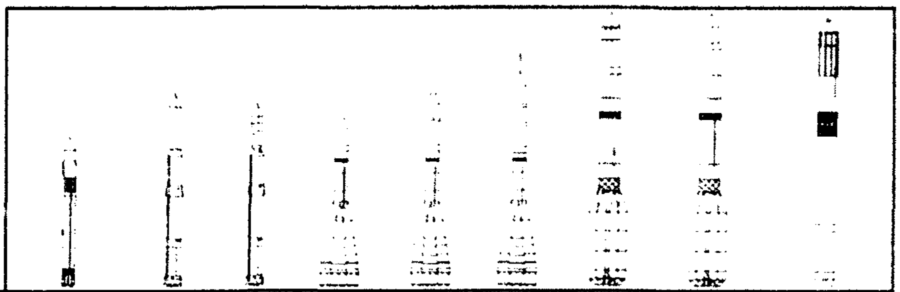
NAME	ARIANE 4	ARIANE 4	ARIANE 4	ARIANE 4	ARIANE 4	ARIANE 4	ARIANE 5
DESIGNATION	ARIANE 40	ARIANE 42P	ARIANE 44P	ARIANE 42L	ARIANE 44LP	ARIANE 44L	ARIANE 5
RESPONSIBLE AGENCY	ARIANESPACE	ARIANESPACE	ARIANESPACE	ARIANESPACE	ARIANESPACE	ARIANESPACE	ARIANESPACE
PRIME CONTRACTOR	CNES	CNES	CNES	CNES	CNES	CNES	CNES
PERFORMANCE -lb (kg)							
LEO POLAR	8,580 (3,900)	10,600 (4,800)	12,100 (5,500)	13,000 (5,900)	14,500 (6,600)	16,900 (7,700)	TBD
LEO DUE EAST	10,800 (4,900)	13,400 (6,100)	15,200 (6,900)	16,300 (7,400)	18,300 (8,300)	21,100 (9,600)	TBD
GTO	4,190 (1,900)	5,730 (2,600)	6,610 (3,000)	7,050 (3,200)	8,160 (3,700)	9,260 (4,200)	15,000 (6,800)
GEO							
RELIABILITY (success rate)	1/1	1/1			6/6	3/4	N/A
PAYLOAD ACCOMMODATION							
DIAMETER – ft (m)	12.0 (3.7)	12.0 (3.7)	12.0 (3.7)	12.0 (3.7)	12.0 (3.7)	12.0 (3.7)	12.0 (3.7)
CYLINDER LENGTH – ft (m)	13.1 (4.0) TO 25.6 (7.8)	13.1 (4.0) TO 25.6 (7.8)	13.1 (4.0) TO 25.6 (7.8)	13.1 (4.0) TO 25.6 (7.8)	13.1 (4.0) TO 25.6 (7.8)	13.1 (4.0) TO 25.6 (7.8)	13.1 (4.0) TO 25.6 (7.8)
CONE LENGTH – ft (m)	15.1 (4.6)	15.1 (4.6)	15.1 (4.6)	15.1 (4.6)	15.1 (4.6)	15.1 (4.6)	15.1 (4.6)
INITIAL LAUNCH CAPABILITY	1990	1990	199X	199X	1988	1989	1995
LAUNCH SITE	ELA-2 KOUROU	ELA-2 KOUROU	ELA-2 KOUROU	ELA-2 KOUROU	ELA-2 KOUROU	ELA-2 KOUROU	ELA-3 KOUROU
LAUNCH CAPABILITY -flt/yr	12	12	12	12	12	12	TBD
RESPONSE TIME	MEDIUM	MEDIUM	MEDIUM	MEDIUM	MEDIUM	MEDIUM	MEDIUM
COST/FLT -$91 M	60-70	62-72	65-75	85-95	90-100	110-120	100-110

Figure 2. Current and in Development European Mainline Launch Systems



Source: SPACE NEWS. December 16-22, 1991

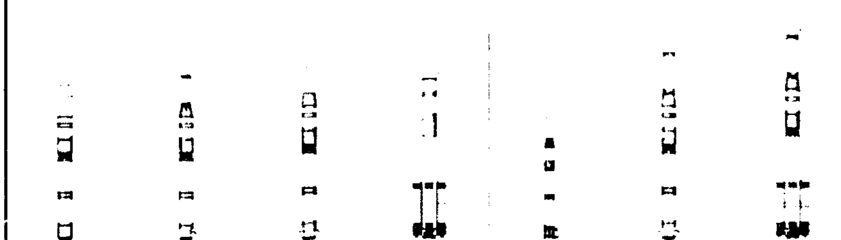
Figure 3. Distribution of Major CIS Space Assets



NAME	KOSMOS	-	TSYKLON	VOSTOK	MOLNIYA	SOYUZ	PROTON	PROTON	ZENIT
DESIGNATION	SL-8	SL-11	SL-14	SL-3	SL-6	SL-4	SL-12	SL-13	SL-16
RESPONSIBLE AGENCY	GLAVKOSMOS		GLAVKOSMOS	GLAVKOSMOS	GLAVKOSMOS	GLAVKOSMOS	GLAVKOSMOS	GLAVKOSMOS	GLAVKOSMOS
PRIME CONTRACTOR	TBD	M. K. YANGEL	M. K. YANGEL	S. P. KOROLEV	S. P. KOROLEV	S. P. KOROLEV	S. P. KOROLEV	S. P. KOROLEV	YUZHNOYE NPO
PERFORMANCE -lb (kg)									
LEO POLAR	-	-	-	-	4 060 (1 840)	-	-	-	25 100 (11 400)
LEO DUE EAST	3 000 (1 350)	-	8 800 (4 000)	10 400 (4 730)	-	15 400 (7 000)	44 100 (20 000)	-	30 300 (13 740)
GTO	-	-	-	-	-	-	-	12 100 (5 500)	12 900 (5 850)
GEO	-	-	-	-	-	-	-	4 850 (2 200)	5 380 (2 440)
RELIABILITY (success rate)	371/377	-	198/201	88/89	179/189	554/566	23/26	138/157	12/14
PAYLOAD ACCOMMODATION									
DIAMETER - ft (m)	7.3 (2.2)	-	7.9 (2.4)	29.3 (8.9)	30.0 (9.1)	37.0 (11.2)	10.9; 12.2 (3.3; 3.7)	10.9; 12.2 (3.3; 3.7)	10.9 (3.3)
CYLINDER LENGTH - ft (m)	5.9 (1.8)	-	11.2 (3.4)	12.9 (3.9)	21.4 (6.5)	22.1 (6.7)			11.5 TO 27.7 (3.5 TO 8.4)
CONE LENGTH - ft (m)	9.9 (3.0)	-	8.2 (2.5)	8.9 (2.7)	8.9 (2.7)	9.2 (2.8)	7.9; 8.6 (2.4; 2.6)	7.9; 8.6 (2.4; 2.6)	12.5 (3.8)
INITIAL LAUNCH CAPABILITY	1964	1967	1977	1959	1961	1963	1967	1968	1985
LAUNCH SITE	PLESETSK, KAPUSTIN YAR, TYURATAM	-	BAIKONUR, PLESETSK	BAIKONUR PLESETSK	BAIKONUR PLESETSK	BAIKONUR PLESETSK	BAIKONUR	BAIKONUR	BAIKONUR CAPE YORK (proposed)
LAUNCH CAPABILITY -flt/yr	7-10	-	10	10	10	50	20	20	10
RESPONSE TIME	LOW	-	MEDIUM	MEDIUM	MEDIUM	MEDIUM	MEDIUM	MEDIUM	MEDIUM
COST/FLT -$91 M	TBD	-	10	14	15	15	35-70	35-70	35-70

Figure 4 CIS Mainline Expendable Launch Systems

CURRENT PRODUCTION IN DEVELOPMENT



NAME	LONG MARCH	LONG MARCH	LONG MARCH	LONG MARCH	LONG MARCH	LONG MARCH	LONG MARCH
DESIGNATION	CZ-2C	CZ-3	CZ-4	CZ-2E	CZ-ID	CZ-3A	CZ-2E/HO
RESPONSIBLE AGENCY	MINISTRY OF ASTRONAUTICS	MINISTRY OF ASTRONAUTICS	MINISTRY OF ASTRONAUTICS	MINISTRY OF ASTRONAUTICS	MINISTRY OF ASTRONAUTICS	MINISTRY OF ASTRONAUTICS	MINISTRY OF ASTRONAUTICS
PRIME CONTRACTOR	CGWIC	CGWIC	CGWIC	CGWIC	CGWIC	CGWIC	CGWIC
PERFORMANCE -lb (kg)							
LEO POLAR	3,860 (1,750)	-	-	-	1,210 (550)	-	-
LEO DUE EAST	7,040 (3,200)	11,000 (5,000)	8,800 (4,000)	20,300 (9,200)	1,630 (740)	15,800 (7,200)	29,900 (13,600)
GTO	2,200 (1,000)	3,300 (1,500)	2,430 (1,110)	7,430 (3,370)	440 (200)	5,500 (2,500)	9,900 (4,500)
GEO	860 (390)	1,600 (730)	1,220 (550)	3,300 (1,500)	220 (100)	2,700 (1,230)	4,950 (2,250)
RELIABILITY (success rate)	TBD	TBD	TBD	TBD	TBD	TBD	TBD
PAYLOAD ACCOMMODATION							
DIAMETER – ft (m)	5.9; 10.2 (1.8; 3.1)	7.6; 8.9 (2.3; 2.7)	7.9; 9.9 (2.4; 3.0)	12.5 (3.8	6.6 (2.0)	9.9; 13.2 (3.0; 4.0)	12.5 (3.8)
CYLINDER LENGTH – ft (m)	0.0; 6.6 (0.0; 2.0)	5.6; 8.6 (1.7; 2.6)	3.3; 9.9 (1.0; 3.0)	20.0 (6.0)	5.3 (1.6)	13.2; 21.4 (4.0; 6.5)	19.8 (6.0)
CONE LENGTH – ft (m)	9.9; 11.5 (3.0; 3.5)	8.2; 9.2 (2.5; 2.8)	7.3; 10.3 (2.2; 3.2)	18.1 (5.5)	7.9 (2.4)	11.2; 16.5 (3.4; 5.0)	18.1 (5.5)
INITIAL LAUNCH CAPABILITY	1975	1984	1988	1990	1991	1992	1995
LAUNCH SITE	JSLC	XSLC	TSLC	XSLC	JSLC	XSLC	XSLC
LAUNCH CAPABILITY -flt/yr	TBD	TBD	TBD	TBD	TBD	TBD	TBD
RESPONSE TIME	HIGH	HIGH	HIGH	HIGH	MEDIUM	HIGH	HIGH
COST/FLT -$91 M	20	33	TBD	40	10	TBD	TBD

Figure 5 PRC Mainline Launch Systems

NLS launch capabilities will be first implemented at the NASA Kennedy Space Flight Center (KSC), using the Shuttle Vehicle Assembly Building (VAB) and Launch Complex 39 (LC-39). Next, NLS will be implemented at Cape Canaveral Air Force Station (CCAFS) with a single launch pad and processing facilities compatible with the Integrated Transport Launch (ITL) concept. NLS evolution planning envisions additional launch capabilities at Canaveral.

In April, 1991, the NLS Program was reviewed and approved by the US National Space Council as a major initiative for fulfilling national space transportation lift capability, operability, and cost reduction needs. Presently, its future is being debated again in Congress.

5.2.2 *Europe*

Europe is currently developing the Ariane 5 as a successor to Ariane 4 but using a completely new design. Its objectives are to deliver into GTO one or more satellites having a total mass of 15,000 lb (6,800 kg) or to deliver the Hermes spaceplane weighing 48,500 lb (22,000 kg) into a 50 x 250 nm (93 x 463 km) transfer orbit at 28.5 deg. Currently the Hermes program is under review.

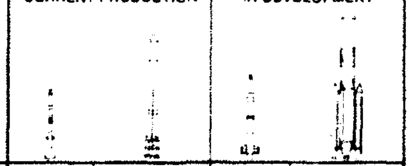
Ariane 5 is an ESA development, but it is intended to satisfy the commercial and non-commercial markets into the next century and will be assigned to Arianespace about a year after first flight, which is predicted for 1995. Following on the success of the Ariane 4, it is conceived as a family of vehicles. A new deployment system, SPELTRA (Structure Porteuse Externe Lancement Triple Ariane) and standardized payload support structures will be available to accommodate single, dual, and triple satellite launches and to accommodate small piggy-back payloads. The capabilities of Ariane 5 are listed in Figure 2.

5.2.3 *Japan*

The first launch of the H-2 vehicle was originally planned for 1992, although problems with the development of the LE-7 LH2/LOX engine will now delay this event. The H-2 is expected to be offered for commercial launches. Its characteristics are shown in Figure 6.

6. SMALL FIXED/RELOCATABLE/ MOBILE LAUNCH SYSTEMS

Small fixed, relocatable, or mobile launch systems are a category of launch systems intended to launch relatively small payloads (and particularly commercial payloads) at low cost. Because their operational procedures are simpler than the mainline systems, their response times tend to be shorter and their launch flexibility to be greater than the mainline systems. Since they have these characteristics, and their payload capabilities are compatible with launching small payloads to low energy orbits, they are strong candidates for performing the low altitude tacsat missions. Figure 7 illustrates two types of orbits that satisfy the low altitude surveillance missions to sun-synchronous or target inclination orbits. Figure 7(a) shows the condition where the satellite is launched from Cape Canaveral at an inclination equal to the latitude of the target area. This has the virtue of passing over the target on the first pass, and, because the orbit is almost parallel to the target latitude, the target will stay in view for several revolutions. Moreover, by placing several satellites in the same inclination, but spaced around the globe from the original orbit, significant view time of the target can be achieved each day. In contrast, Figure 7(b) shows near polar orbits (sun-synchronous) where each satellite overpasses every target on the globe twice per day. By having several satellites in orbits with the same inclination but spaced apart, a target can be viewed in daylight at several different times a day.



	CURRENT PRODUCTION		IN DEVELOPMENT	
NAME	M-VEHICLE	H-VEHICLE	M-VEHICLE	H-VEHICLE
DESIGNATION	M-3S-II	H-1	M-V	H-2
RESPONSIBLE AGENCY	ISAS	NASDA	ISAS	NASDA
PRIME CONTRACTOR	NISSAN	MITSUBISHI HEAVY INDUSTRIES	NISSAN	MITSUBISHI HEAVY INDUSTRIES
PERFORMANCE - lb (kg)				
LEO POLAR	1,300 (590)	4,800 (2,200)	2,680 (1,215)	14,500 (6,600)
LEO DUE EAST	1,720 (780)	7,000 (3,200)	4,300 (1,950)	23,000 (10,500)
GTO	1,140 (517)	2,400 (1,100)	2,680 (1,215)	8,800 (4,000)
GEO	440 (200)	1,200 (550)	1,080 (490)	4,800 (2,200)
RELIABILITY (success rate)	5/5	7/7	N/A	N/A
PAYLOAD ACCOMMODATION				
DIAMETER - ft (m)	3.1 (1.4)	7.2 (2.4)	5.0 (2.2)	12.1 (3.7); 15.1 (4.6)
CYLINDER LENGTH - ft (m)	4.4 (2.0)	8.8 (2.7; 15.2 (4.6)	7.3 (3.3)	20.0 (6.1)
CONE LENGTH - ft (m)	4.4 (2.0)	6.3 (1.9)	4.4 (2.0)	14.4 (4.4)
INITIAL LAUNCH CAPABILITY	1985	1986	1993	1992
LAUNCH SITE	KSC, MLC	OSAKI TANEGASHIMA		YOSHINOBU TANEGASHIMA
LAUNCH CAPABILITY - flt/yr	1	2	1	4
RESPONSE TIME				
COST/FLT - $91 M	31		40	

Figure 6. Current and Planned Japanese Launch Systems

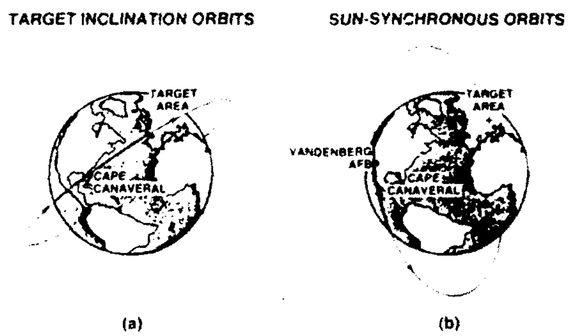


Figure 7. Low Altitude Surveillance Mission Orbits

6.1 Current and Near-Term, Planned and Proposed

6.1.1 *United States (US)*

Table 2 shows the characteristics of a number of small fixed or relocatable US launch vehicles that exist or are in development. The Scout I, which, in any case, is currently out of production, can only deliver 400 to 600 lb (181 to 273 kg) payloads to the low Earth orbits of interest. The Scout II and the Pegasus air-launched booster have approximately the same payload capability to LEO in the 600 to 900 lb (273 to 409 kg) range. The Pegasus has the advantage of flexible launch point location and all-azimuth capability. The Conestoga family of space launchers provides a range of payload capabilities by adding strap-on boosters to the basic four-stage core vehicle. It should be pointed out that the only configuration that has flown is Conestoga I, which successfully flew a sub-orbital mission in 1980. The Conestoga IIA version offers 2 to 3 times the payload capability of the Scout I at the same price ($10–12M), according to the manufacturer. The Taurus, which is under development by Orbital Sciences Corp (OSC) for the Defense Advanced Research and Planning Organization (DARPA), provides considerable payload capability to LEO. In many cases only one satellite per ring is required, but several rings properly spaced to achieve extended coverage may be needed. If this is the case, the additional capability of the Conestoga or the Taurus may be used to permit carrying excess propellant in, say, one of two satellites to boost it to a higher altitude and different inclination in order to generate a differential nodal regression between the two satellite planes. Nodal regression of a satellite plane is due to forces exerted on the satellite caused by the Earth's oblateness. These forces tend to make

Table 2. Current and Planned Small US Fixed/Relocatable/Mobile Launch Systems

LAUNCH VEHICLE	SCOUT 1	SCOUT 2	PEGASUS	PEGASUS XL	TAURUS	CONESTOGA IIA	CONESTOGA IIIA
RESPONSIBLE AGENCY	NASA	NASA	DARPA	USAF/NASA	DARPA	NASA	NASA
PRIME CONTRACTOR	LTV	LTV	OSC	OSC	OSC	EER	EER
PERFORMANCE (lb)							
LEO DUE EAST							
100 nm CIRC	600	1160	740	940	2800	1400	1800
300 nm CIRC	500	925	540	760	2500	1250	1500
LEO POLAR							
100 nm CIRC	460	820	560	720	2300	1200	1550
300 nm CIRC	400	750	390	570	1900	1050	1350
GTO	120	240			620-1100		
RELIABILITY -%	95	-	-	-	-	-	-
PAYLOAD ACCOMMODATION							
DIAMETER - ft	2.5; 3.2	2.5 TO 2.8	3.8	3.8	5.0	4.1	4.1
CYLINDER LENGTH - ft	2.8; 4.0	2.7 TO 4.0	3.0	3.9	8.0	6.9	8.9
CONE LENGTH - ft	1.3; 2.5	1.2 TO 3.5	3.3	3.3	9.0	3.3	3.3
INITIAL LAUNCH CAPABILITY	1960 (out of production)	1990	1990	1993	1993	*20 mo FROM GO-AHEAD	*20 mo FROM GO-AHEAD
COST/FLIGHT - $91 M	10-12	15	6	13 – 14	15 +	10 – 12	12 – 15

*Conestoga I flew suborbital in 1980

the orbital plane effectively precess about the polar axis of the Earth. When the plane of the second satellite has achieved the desired separation from that of the first, a second burn of that satellite's propulsion can be used to return the satellite to the same altitude and inclination as the other. In this case, the excess boost capability of the Taurus might be used to launch a silent spare. It should be pointed out that the Taurus system is being designed to be relocatable for survivability reasons since it was proposed for Space Defense Initiative (SDI) use. The SDI-postulated requirement for the system was to be able to establish a launch site in five days and launch a satellite three days later. A convoy of trucks carrying the appropriate equipment provides this capability. Further study is required to determine if this capability is of any benefit to the tacsat mission.

Scout 1. The NASA Scout vehicle became operational in 1960; as of July 1991 only four vehicles remained. It is a four-stage, solid propellant, series burn rocket. The Scout can deliver 146 kg to a sun-synchronous polar orbit, 460 lb (220 kg) to a LEO polar orbit (launched from Vandenberg), or 570 lb (259 kg) to an easterly (37.7 deg) orbit launched from Wallops Island, Virginia. The Scout is essentially phased out as a US launch vehicle although it is anticipated that the Italians will support an enhanced (Scout 2) program in cooperation with US industry. The Italians will launch from San Marco, off the coast of Kenya.

The minimum Scout launch vehicle-only cost is quoted at $10M, which increases to $12-13M for full launch service. This translates to $25,000 to $37,300 per lb ($55,00 to $82,000 per kg), which does not compare favorably with the current industry average of $10,000 to $11,800 per lb ($22,000 to $26,000 per kg) for delivering large commercial satellites to GTO.

Scout 2. An upgraded version (Scout 2) has been studied that can double the payload capability of Scout 1. Strap-on solids and an apogee kick motor added to the existing core vehicle yield the enhanced performance.

Taurus. Taurus is a DARPA development of a Standard Small Launch Vehicle (SSLV) based on the Pegasus with the addition of a Peacekeeper first stage. It will deliver 3,000 lb (1,364 kg) to LEO or 800 lb (364 kg) to GEO. Taurus will be ground-launched and will demonstrate the rapid establishment of a ground mobile launch capability. A response time of 72 hours from alert to launch is the goal.

Conestoga. This is a commercial development of Space Services Incorporated (SSI), which is now a division of EER. A NASA-funded effort entitled the Commercial Experiment Transporter (COMET) using commercial business practices is underway at Westinghouse Electric Corporation to demonstrate the economical development of a reliable launch and payload recovery system. SSI is providing the launch services segment of the program and is using rocket technology developed by Israeli Aircraft Industry (IAI), rather than surplus US government motors, to increase reliability and reduce insurance costs. The family of vehicles is designed for evolutionary growth.

Conestoga II delivers a payload of 700 lb (318 kg) to 250 nm (463 km) polar orbit, or 400 lb (182 kg) to 400 nm (740 km) polar orbit. Conestoga IV delivers payloads of 2,000 lb (909 kg) and 1500 lb (682 kg), respectively. The vehicle family is designed to use a relocatable launch site. Launch is nominally from Wallops Flight Facility, VAFB, or White Sands Missile Range. In addition, the vehicles are also designed to be compatible with proposed Hawaii, Florida, or Cape York spaceports.

Orbital Express. A contract has been awarded to International MicroSpace, Inc. for launch services for the SDIO's Miniature Seeker Technology Integration (MSTI) program. International MicroSpace, of Herndon, Virginia, is offering its Orbital Express ground-based launcher to deliver a payload of 400 lb (182 kg) into a sun-synchronous orbit. The Orbital Express has a Castor IVb first stage, Castor I second stage, Star 31 third stage, and a Star 20 fourth stage. It is expected to be operational in late 1993, and is one member of a planned family of vehicles.

Pegasus. The US Pegasus is the only system currently available that provides a mobile, all-azimuth launch platform and therefore has the potential to satisfy any launch flexibility requirements of the tacsat mission. The development program is a privately funded joint venture by Orbital Sciences Corporation (OSC) and Hercules Aerospace Company. Pegasus is a three-stage, solid-propellant, inertially guided, all composite, winged launch vehicle based on Trident and Pershing motor technology. Flights that have been made so far have utilized a Boeing B-52 to level-flight conditions at approximately 40,000 ft (12,200 m) and a speed of Mach 0.8. After release from the aircraft and ignition of the Stage I motor, the autonomous flight control system provides all the guidance necessary to produce a wide range of suborbital and orbital trajectories. The development program was begun in early 1987, and the vehicle has been available for launch services to both government and commercial users beginning with its first orbital flight in April 1990. The second flight of the Pegasus featured a new operational propulsion unit and a guidance upgrade. After burnout of the third solid-fuel stage, a small liquid propulsion unit, dubbed HAPS (Hydrazine Auxiliary Propulsion System), functions as a precision orbital injection kit. In addition, a Global Positioning System (GPS) receiver serves as a redundant source of navigation information, while the Inertial Measurement Unit (IMU) serves as the primary source. The current configuration, when air-launched from a B-52 bomber, delivers 750 to 1,000 lb (341 to 454 kg) of payload to a due east orbit and 560 to 750 lb (255 to 341 kg) to a LEO polar orbit. Improvements have been identified to increase the payload capability to 2,500 to 3,100 lb (1136 to 1409 kg) to due east and 1,880 to 2,300 lb (818 to 1045 kg)to LEO polar orbits. A Lockheed L-1011 is being modified to serve as a commercial launch platform. Currently Pegasus gets 95% of its revenue from government contracting. Cost is about $6.5M per flight to LEO. Commercial flights are being offered at about $4.5M per flight to LEO.

LTV Standard Small Launch Vehicle (SSLV). The LTV (Ling/Tempco/Vought) Standard Small Launch Vehicle (SSLV) is intended to be a transportable, ground-launched vehicle with the ability to be rapidly deployed. The response time goal, from alert to launch, is 72 hours. Payload requirements range between the Pegasus and Delta class vehicles (1,000 lb (455 kg) into a 400 nm (740 km) polar orbit) and the capability to launch multiple lightsats into LEO or single lightsats into a variety of highly elliptical orbits, which can maximize time over a particular theater of operations (4, 6, 8, or 12 hour Molniya orbits). The SSLV proposes to utilize Minuteman, Polaris, or Poseidon stages.

The future availability of this family of vehicles is unsure since LTV is bankrupt and the LTV missile part of the company is being auctioned to the highest bidder. At this time, a bid from a group led by Loral is favored.

SEALAR. The Naval Research Laboratory (NRL) Sea Launch and Recovery (SEALAR) is conceived as a two-stage liquid-fueled launch vehicle that would take off from the ocean surface. The first stage would fall back into the ocean and be recovered. Its design goal is to lift 10,000 lb (4,545 kg) into LEO for $500 per lb ($1,100 per kg). The Navy has drop-tested a prototype built by Truax Engineering, Saratoga, California. To date the SEALAR project has been supported by limited Navy research and development and SDIO funding.

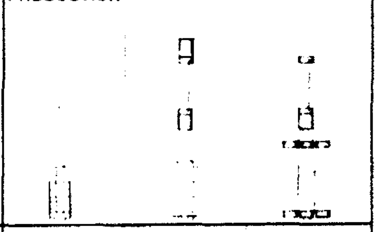
Others. There are other US companies attempting to market small low-cost launch systems, including the American Rocket Company (AMROC), which is proposing the hybrid Acquila Industrial Launch Vehicle (ILV); Lockheed Missile and Space, which has proposed vehicles based on Sea Launched Ballistic Missile (SLBM) technology; Pacific American Launch Systems, which is proposing the Liberty IA vehicle; and E'Prime, which is proposing the Eagle launch vehicle for deploying the commercial Iridium satellite constellation. E'Prime is also promoting the "Unified Satellite Transfer Module (USTM)," which exploits the concept of a standardized satellite bus integrated as an intrinsic part of the launch vehicle development. OSC is also promoting a similar concept as part of the Pegasus program. The concept of a standardized bus may be a way of reducing the cost of the tacsat mission. Examples of microbuses in the commercial sector that are actually in production include: the European Space Agency (ESA) Ariane Technology Experiment Platform (ArTEP), the French Matra Janus, the Italian Italspazio Microsat, and the British University of Surrey UoSat, built by Surrey Satellite Technology Ltd.

6.1.2 *Japan*

M-V Series. The first of the M Family of vehicles were either three- or four-stage all solid-propellant vehicles made use of aerodynamic and spin-stabilization control techniques. Later versions used secondary fuel injection for thrust vector control in response to an on-board autopilot, and eventually the control system was digitized to increase control logic design flexibility and to reduce power and weight needs. The current production version is the M-3S-II, whose characteristics are listed in Figure 6.

6.1.3 *India*

SLV Series. The Augmented Space Launch Vehicle (ASLV) can lift 330 lb (150 kg) into LEO. The launch of the Polar Satellite Launch Vehicle (PSLV), planned for March, 1993, is expected to deliver 6,600 lb (3,000 kg) to LEO. The PSLV, unlike the ASLV, which is composed of four solid stages and two strap-ons, will use liquid second and fourth stages. The US has placed sanctions on the Indian Space Research Organization (ISRO) for buying advanced rocket technology from Russia. The characteristics of the Indian vehicles are listed in Figure 8.



	CURRENT PRODUCTION	IN DEVELOPMENT	
NAME			
DESIGNATION	ASLV	PSLV	GSLV
RESPONSIBLE AGENCY	ISRO	ISRO	ISRO
PRIME CONTRACTOR	HINDUSTAN AERONAUTICS	HINDUSTAN AERONAUTICS	HINDUSTAN AERONAUTICS
PERFORMANCE -lb (kg)			
LEO POLAR	-	2,200 (1,000)	-
LEO DUE EAST	330 (150)	6,600 (3,000)	17,600 (8,000)
GTO	-	990 (450)	5,500 (2,500)
GEO	-	-	-
RELIABILITY (success rate)	1/3	N/A	N/A
PAYLOAD ACCOMMODATION			
DIAMETER - ft (m)	3.3 (1.0)	10.5 (3.2)	TBD
CYLINDER LENGTH - ft (m)	9.8 (3.0)	27.0 (8.3)	TBD
CONE LENGTH - ft (m)	-	-	TBD
INITIAL LAUNCH CAPABILITY	1987	1992	1995-96
LAUNCH SITE	SRIHARIKOTA	SRIHARIKOTA	SRIHARIKOTA
LAUNCH CAPABILITY -flt/yr	1-2	1-2	1-2
RESPONSE TIME	MEDIUM	MEDIUM	MEDIUM
COST/FLT - $91 M	TBD	TBD	TBD

Figure 8. Current and Planned Indian Launch Systems

6.1.4 *Israel*

Shavit. The Shavit (which translates to Comet) is a three-stage solid propellant vehicle and is a modification of the Jericho II Intermediate Range Ballistic Missile (IRBM). The Jericho is claimed to be capable of launching a 660 lb (300 kg) warhead. A satellite, Offeq 2, was launched successfully on 1 April, 1990 into an elliptical orbit with a perigee of 126 nm (210 km) and an apogee of 900 nm (1,500 km). The satellite weight is estimated to be 352 lb (160 kg). The LEO capacity is an estimated 440 lb (200 kg) into a retrograde orbit.

6.1.5 *Brazil*

VLS. The VLS launch system is a conventional solid propellant four-stage system developed from the family of Sonda sounding rockets. It is designed to deliver 220 to 440 lb (100 to 200 kg) payloads into circular orbits ranging from 135 to 540 nm (250 to 1,000 km) at various inclinations. The nominal payload is 150 kg into a 750 km circular equatorial orbit.

6.2 Future Planned and Proposed

6.2.1 *Australia*

A consortium of Australian companies led by British Aerospace, Australia, Ltd is engaged in supporting a joint venture between Auspace Pty Ltd and Hawker de Havilland Ltd to develop the Southern Launch Vehicle (SLV) (sometimes referred to as the Australian Launch Vehicle). The SLV is seen as a carrier of small satellites into LEO with payloads up to 1.5 tonnes. The Association of Australian Aerospace Industries has also suggested the development of a lightsat platform based on the proven British UoSat microsatellite technology. Initial launches would be made from Woomera, but if the Cape York facility is developed, launches could take place from there too. Eight to 10 launches per year are projected.

6.2.2 *United Kingdom/Norway*

Little Launcher for Low Earth Orbit (LittLEO). The commercial LittLEO program is being undertaken by General Technology Systems Ltd and the Norwegian Space Center (Norsk Romsenter) to offer a low-cost and reliable means of placing small payloads into both polar and sun-synchronous orbits from a European launch site. The LittLEO is a solid propellant vehicle capable of placing a 1,320 lb (600 kg) payload into a 243 nm (450 km) polar orbit or 1,547 lb (703 kg) into a 162 nm (300 km) polar orbit. It would be launched from the Andoya Rocket Range (69 deg 17 min N, 16 deg 01 min E). More than 400 sounding rockets have been launched from this site. LittLEO could also launch 2,006 lb (912 kg) into a 162 nm (300 km) polar orbit from Italy's San Marco launch site.

6.2.3 *Spain*

Capricornio. The Instituto Nacional De Tecnica Aerospacial (INTA) has announced a new rocket development, the Capricornio, which is intended to provide low-cost space transportation for the scientific and communications community. It is a three-stage all-solid rocket capable of placing 110 to 220 lb (50 to 100 kg) into a 324 nm (600 km) polar orbit. Tests will be conducted from El Arenosillo (Huelva), and there are plans to build a future launch complex in the Canary Islands.

6.2.4 *Commonwealth of Independent States (CIS)*

Space Clipper. Yuzhnoye NPO, based in the Ukraine, is proposing an air-launched commercial launch system, spaceclipper, which is planned to be available in 1994. The system consists of an An-124 carrier aircraft and a choice of several three- or four-stage solid propellant launch vehicles (derived from the SS-24 missile) that are launched from the cargo bay of the An-124. A choice of six different solid rocket launchers has been proposed. The baseline system is capable of delivering 1,100 lb (500 kg) to LEO and is modularized to provide six different versions of the vehicle and adapt to a wide variety of orbital requirements. The maximum takeoff weight is 392 tonnes, and the system is claimed to be able to operate from any airfield capable of servicing a Boeing 747. To avoid technology transfer restrictions, the installation of the payload can be performed on-board the Antonov at the customer's airfield. Yuzhnoye plans to modify two An-124 aircraft, and there are more than 100 SS-24s in inventory. It is claimed that the combustion products of the Space Clipper are non-polluting. Yuzhnoye also builds the Zenit commercial launcher. The Space Clipper concept is illustrated in Figure 9.

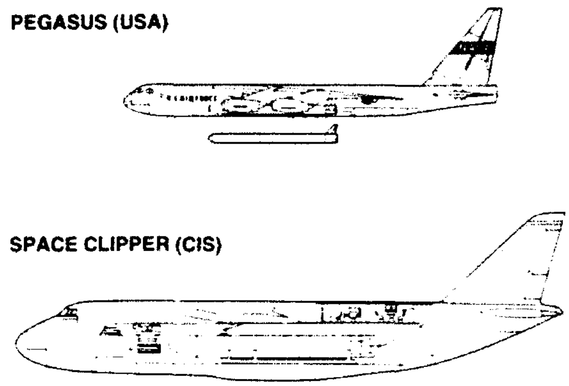


Figure 9. Air-Borne Mobile Launch Vehicles

7. BALLISTIC MISSILE-DERIVED LAUNCH SYSTEMS

As a result of the US/CIS Strategic Arms Reduction Talks (START) agreements, the use of surplus strategic ballistic missiles for launching tacsats needs to be considered. There are, of course, many issues that must be addressed in determining the practicality and advantages of this option. The factors include the latest trends in the arms reduction talks in terms of determining the types and numbers of strategic ballistic missiles that might become available and any treaty restrictions on their use as space launchers; the payload capability to typical orbits of interest and the payload compartment dimensions; and estimates of the non-recurring and recurring costs. With respect to launch responsiveness, the land-based ballistic missile on alert status represents the ultimate, with response times in the order of seconds. The cost and other issues related to maintaining this type of capability presents the limiting case of launch responsiveness.

The trend in the strategic nuclear warhead reduction talks is for both the US and the CIS to go in two phases from the current total count for all three legs of the Triad of somewhat more than 10,000 warheads to 3,000-3,500 warheads by the year 2003. The first phase reduces the total number of warheads to 3,800-4,250 in seven years or less from the execution date of the treaty. Within these overall Triad limits, there are specific limits placed on the Intercontinental Ballistic Missile (ICBM) and Sea Launched Ballistic Missile (SLBM) forces. At the end of the first phase, the number of ICBMs cannot exceed 1,200, and the number of SLBMs cannot exceed 2160; for the second phase these limits are 1,200 and 1,750, respectively. In the second phase there is also a requirement that all the remaining ICBMs carry only one warhead each, while the SLBMs can continue to carry multiple warheads.

7.1 United States

Within the limits outlined above, there are still a large number of possible distributions of specific missile reductions and therefore of their availability for space launch. However, as far as US missiles are concerned, Table 3 shows a generally accepted retirement schedule and indicates that, between now and the year 2000-2003, the Minuteman IIs (MM-IIs) and the Navy Poseidon C-3 and Trident C-4 missiles will be retired and may be available for other applications. After that time period, the Treaty will require Peacekeeper missiles to be retired because of Multiple Independently-Targeted Reentry Vehicle (MIRV) limitations and specific de-MIRVing rules. The exact number of Peacekeepers that would be

Table 3. US Ballistic Missile Retirement Schedule

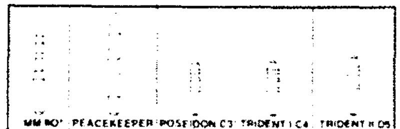
- **MINUTEMAN II**
  - **BEGAN ^ANSFERRING BETWEEN 20 AND 30 THIRD STAGES TC STORAGE IN 1991**
  - **450 WILL EVENTUALLY BE RETIRED**
  - **COMMERCIAL THIRD STAGE (Aerojet Orbus) GIVES MM-II-DERIVATIVE SAME PERFORMANCE AS MM-II**
- **PEACEKEEPER**
  - **RECENT DEVELOPMENT**
  - **50 MISSILES MAY BE RETIRED IN 1995**
- **POSEIDON C3**
  - **NAVY WILL MAKE 100 AVAILABLE TO AF/BMO**
  - **REMAINDER TO BE DESTROYED**
- **TRIDENT C4**
  - **NAVY WILL RETAIN INDEFINITELY**

available then will depend on Congressional decisions for additional buys, but the number will be at least the fifty currently deployed.

Regarding any constraints on the use of these assets, it is particularly noteworthy that in contrast to the International Nuclear Force (INF) (Theater) Tactical Nuclear Missile Limitation Treaty, which requires each side to destroy these missiles, the START treaty will not only not require that the retired missiles be destroyed, but will explicitly permit their use as space launchers subject to certain conditions. These conditions include the following:

- The vehicles must be launched from existing test ranges or space launch facilities, including up to no more than seven new launch pads at the existing sites.
- The vehicles cannot be launched from mobile launchers.
- The vehicles cannot be launched from an airplane or any waterborne platform, other than a submarine.
- Telemetry must be transmitted unencrypted, and a tape supplied to the other party to the treaty, except for 11 missions per year using fully retired vehicles that may return encrypted data.

Figure 10 shows the performance of the ballistic missile derivatives to four typical orbits.. Although it currently does not appear likely that the MM-III fleet will be retired, MM-III data is presented because the commercially available Orbus third stage converts MM-II performance to that of MM-III. It is seen that the MM-II and the Poseidon C3 have fairly limited performance, while the MM-II/Orbus has capability that is closer to the tacsat range of interest. The Peacekeeper has fairly significant capability but may not be available for other applications until 2003 and beyond with the possible exception of the 50 missiles mentioned earlier. The volume available for the payload is somewhat smaller than that of other small launch vehicles, but preliminary in-house Aerospace studies of small Peacekeeper-launched satellites with no modification to the external dimensions of the launch vehicle (so it could still be launched from a silo if survivability is an issue) indicate it is feasible to package viable satellites. For cases where survivability is not an issue, the payload fairing can be enlarged to provide more room at the expense of some performance. Other factors that must be studied in more detail include the feasibility of designing satellites to withstand the more severe launch environment produced by ballistic missiles, that is, the 11 g's acceleration of Peacekeeper. Again, very preliminary estimates indicate this should not be a problem. Finally, the costs, which have been provided by the Air Force Multi-Service Launch Systems (MSLS) organization, are seen to be considerably lower than those of the Pegasus and Scout II vehicles presented earlier on a dollars per pound to orbit basis, and about the



	MM IIO*	PEACEKEEPER	POSEIDON C3	TRIDENT I C4	TRIDENT II D5
RESPONSIBLE AGENCY	USAF	USAF	US NAVY	US NAVY	US NAVY
PERFORMANCE - lb					NOT AVAILABLE
• LEO DUE EAST					
100 nm CIRC	1190	4850	800		
300 nm CIRC	1010	3150	725		
• LEO POLAR					
100 nm CIRC	850	3900	650	1000	
300 nm CIRC	710		600		
RELIABILITY - %	>0.95	-	0.9		
PAYLOAD ACCOMMODATIONS					
• DIAMETER - ft	4.33	7.67	6.0		
• CYL LENGTH + CONE LENGTH - ft	12.0	12.0	4.2		
INITIAL LAUNCH CAPABILITY	1995				
COST/FLIGHT $91 M	5.6**	6.0	~6.0	?	

* MM IIO is MM II with 3rd stage replaced by commercial Orbus 7 motor and a Star 37 FM 4th stage
** In addition, there may be a first time cost of $4.1 M

Figure 10. US Ballistic Missile-Derived Launch Vehicle Performance Characteristics

same as the Taurus. Once again, controlling satellite weight and size is seen to provide large dividends in lower launch costs.

An option that has not been checked for viability or cost is to launch tacsats from a submarine at the South Pole. The satellite could be in orbit before it crossed the equator. In the case of SLBMs the treaty mandates: (a) retiring submarines; (b) reducing the number of warheads per tube; or (c) sailing with empty tubes. If the empty-tube option were selected, the empty tubes could perhaps be used for tacsat launches.

Minuteman Derivatives. Martin Marietta Launch Systems, Denver, Colorado, has been awarded a $133M Air Force contract to modify 44 MM-IIs for space launch. The basic contract is for two launches at $30M, with options for 42 more. The MM-II has three stages. Martin Marietta will add a fourth stage (the Orbus, built by Aerojet), bringing its capability up to approximately the same as MM-III, and a new guidance system that would govern the entire booster. First launch is expected from VAFB in 1994. Launch costs are expected to be about $6M to $8M per launch, with a future reduction to $4M per launch.

Polaris/Trident/Poseidon Derivatives. Lockheed Missiles and Space Company (LMSC) has proposed using the Poseidon C-3 submarine missile as a small space launch vehicle that could deliver 700 lb (318 kg) to a 270 x 270 nm (500 x 500 km) 70 deg orbit. There are about 60 C-3 vehicles in storage. Other options include the Trident I C-4 and the Trident II D-5. The C-4 could deliver 1200 lb (545 kg) to LEO; the D-5 could deliver 2000 lb (909 kg). It should be noted that, at this time, it does not appear that the US Navy will make submarine missiles available for conversion and, in fact, has plans to destroy any surplus hardware resulting from treaty negotiations.

7.2 Commonwealth of Independent States (CIS)

The CIS has surplus ballistic missile hardware it would like to exchange for hard currency. The ex-Soviet space organization, Glavkosmos, has reorganized to market joint Russian-Kazakhistan space launch services (cooperation is needed since the Baikonur Cosmodrome is located in Kazakhstan, which has formed its own national space agency) and has test flown a converted SS-19 missile; there are about 300 SS-19 missiles in inventory, each capable of carrying six nuclear warheads. The smaller SS-25 can carry only a single warhead, but there are more than 300 in the military's inventory. The world's most powerful ICBM, the SS-18, has also been proposed as a space launch vehicle. This vehicle can launch up to a dozen small (100–200 lb or 45–91 kg) satellites into orbit for less than $30 and is claimed to have all-weather capability. There are estimated to be more than 300 SS-18 missiles in the CIS. START calls for a 100% reduction in that force.

There is concern on the part of the US commercial space launch industry that the ex-Soviet surplus space hardware will be dumped on the world market. This may justify a US agreement to convert the ex-Soviet surplus ballistic missiles that are not destined for destruction into space launchers that are held in reserve and only launched in the common interest of world peace, in response to critical theater warfare needs.

7.3 Europe

7.3.1 *United Kingdom (UK)*

Malcolm Rifkind, a defence secretary in the British Ministry of Defence, announced on June 15, 1992, that the only nuclear weapons Britain will retain will be a handful of WE-177 gravity bombs (as part of its commitment to NATO) and its strategic Polaris missiles. There is a debate over what, if anything, should be done about the Polaris-replacing Trident program that is scheduled to carry 128 nuclear warheads on 16 Trident missiles. If the Trident program is continued as presently scheduled, it is possible that some surplus British Polaris missile components will become available, although they will probably become part of the US inventory.

7.3.2 *France*

France has not released a policy statement on the future of any French surplus ballistic missiles.

7.4 Others

It is possible that both China and the CIS could, in the future, market ballistic missiles to developing nations. The future will probably see a proliferation of ballistic missiles throughout the world. China, for instance, has a large inventory, which it has shown no inclination to reduce. Other smaller countries that feel threatened by their neighbors are more likely to add to, rather than reduce, their inventory of ballistic missiles. The wide availability of these weapons could encourage the rise of dangerous despotic regimes anywhere in the world, and establishes a strong case for conducting detailed tacsat studies.

8. FAR-TERM LAUNCH SYSTEMS

The emphasis of this paper is on current or near-term launch system options. However, tacsat studies also need to look into the next century, when the need for such systems may intensify. A number of advanced systems are being investigated that may have application to future tacsat missions.

8.1 United States

8.1.1 *Single-Stage-To-Orbit (SSTO)*

A single-stage-to-orbit vehicle has long been the desire of many space transportation planners because of its potential for reducing operational complexity and cost, and providing launch-on-demand capability for critical military systems. The SDIO has examined a number of SSTO configurations to satisfy postulated SDIO mission launch requirements, and selected the McDonnell Douglas Delta Clipper for further study. The Single-Stage-Rocket Technology (SSRT) Program is now underway at McDonnell Douglas to validate some of the critical technologies. A sub-orbital flight from White Sands is planned for 1993.

The generic SSTO concept has some technological aspects in common with the National Aerospace Plane (NASP) program described below. However, the long-term NASP effort (the X-30 program) is focused on building an air-breathing SSTO technology demonstrator, with plans for a NASP-derived operational vehicle, whereas the SSTO effort would be focused on building a prototype operational rocket-powered SSTO vehicle.

8.1.2 *Two-Stage-To-Orbit (TSTO)*

Several US contractors have presented plans for developing a Two-Stage-To-Orbit (TSTO) system where either one or both stages are fully reusable. Although such concepts have been under study since the early 1960s, they are presently only contractor Independent Research and Development (IR&D) programs in the very early concept exploration phase of acquisition.

8.1.3 *National AeroSpace Plane (NASP)*

The NASP Program has the overall goal of providing the technological basis for future hypersonic flight vehicles for application to both civilian and military systems. Currently, because of budget constraints, funding is being focused on power plant technology. The NASP propulsion systems are to be largely air-breathing, consisting of combined ramjet-scramjet engines fueled by slush hydrogen. The X-30 is not a prototype of any specific military or civilian system but will be used to demonstrate the requisite technology for such future systems, including airplanes with hypersonic cruise capability and fully reusable launch vehicles for payload delivery to orbit. These vehicles will use conventional runways for takeoff and landing. A National AeroSpace Plane (NASP) Derived Vehicle (NDV) would be an operational follow-on to NASP. An Interim NASP that would fly operational sub-orbital advanced tacsat missions has been suggested as a candidate for supporting tacsat missions that could evolve in the next century. The NASP concept is illustrated in Figure 11.

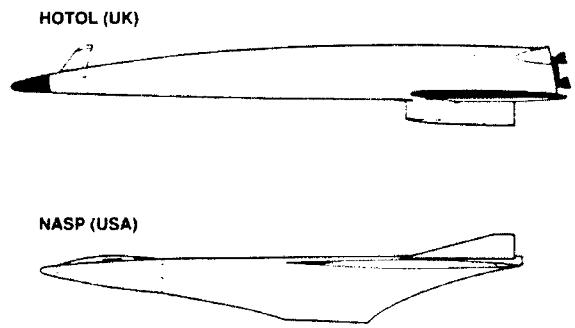


Figure 11. Representative Horizontal Take-off Single Stage Launch Vehicles

8.2 Europe

Several European nations are investigating advanced fully or partially reusable launch vehicles that may eventually need to be considered as space launchers for tacsat missions, the major ones being France, Germany, and the United Kingdom.

8.2.1 *Germany*

Sanger. The Sanger is a reference concept for the German Hypersonics Technology Program. Currently the vehicle concept is a two-stage fully reusable system. The program was initiated in 1988 and will be executed in several consecutive phases. The current Phase 1 is planned to continue until the end of 1992, with a possible extension of two to three years. The results of the program so far have confirmed the reference Sanger concept theoretically and experimentally. Only one change has been made to replace the expendable cargo version of the second stage by a reusable unmanned version. The program is increasing the participation of international partners, which should lead to a program proposal to ESA. Benefits of the Sanger concept are that it would provide a launch capability from continental Europe, and its mobility would permit it to take advantage of equatorial launch of the final stage. The Sanger concept is illustrated in Figure 12.

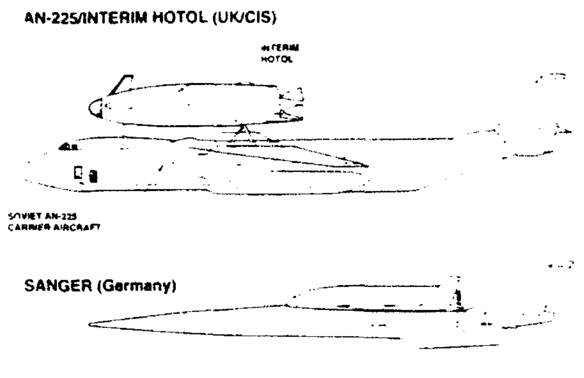


Figure 12. Representative Horizontal Take-off Two-Stage Vehicles

8.2.2 *United Kingdom*

Horizontal Take-Off and Landing (HOTOL). The original ambitious HOTOL program is currently unfunded. The engine concept for the HOTOL developed and patented by Alan Bond has been declassified, but the Rolls-Royce RB5454 design is still held proprietary. The engine is a pre-cooled, air-breathing rocket that utilizes air entering through a V-wedged intake. The air is shocked down to subsonic speeds and passed through a series of heat exchangers and turbocompressors until it reaches rocket chamber pressures. Liquid hydrogen and liquified air fuel the first phase of engine flight up to Mach 5.5 at an altitude of 98,430 ft (30 km). Onboard liquid oxygen and hydrogen is used for the remainder of the flight. HOTOL shares technology needs with the US NASP but its future is uncertain. It is illustrated in Figure 11.

8.3 United Kingdom/Russia

Interim HOTOL. A joint British/Russian development to fly an interim version of the British HOTOL on the Russian Antonov-225 "MPIR" heavy lift aircraft is underway. Successful wind-tunnel tests at simulated speeds of Mach 10.5 and 14 have been conducted at the Central Aerohydrodynamics Institute (TsAGI) in their T-128 wind tunnel in Moscow. The three organizations have signed agreements among themselves as well as with their respective governments. British Aerospace is defining the Aerospaceplane and airborne support systems. Limited resources are available, but the European Community Commission has been approached by British Aerospace for support since the commission has said it would entertain joint European/Soviet (now CIS) aerospace ventures.

The vehicle is planned to carry a payload of 15,400 to 17,600 lb (7,000 to 8,000 kg) into LEO after being launched from the An-225 at an altitude of about 29,500 ft (9 km). The An-225 requires the addition of two extra Soviet D-18 engines. Rolls Royce replacement engines were considered to avoid installing additional engines but was rejected as not a cost-effective modification. The concept is illustrated in Figure 12.

8.4 Others

The ex-Soviet Union has conducted advanced partially and fully reusable vehicle research and testing for many years. In November, 1991, for instance, the Central Institute of Aviation Motors, Moscow, claims to have conducted the first test flight of a hydrogen-fueled scramjet engine. The test flight was of a missile-like vehicle powered by a conventional rocket engine to an altitude of about 16 nm (30 km). A ramjet engine then ignited and accelerated the vehicle to a speed of Mach 6. During the flight, supersonic combustion occurred. Attempts are being made to market ex-Soviet resources to other nations, such as the US, France, Germany, and Japan. Several other nations, including France, Japan, and India have examined advanced reusable vehicles, both two-stage and single-stage. The world-wide decreasing space budgets will make international cooperative ventures most likely in the future.

9. LAUNCH SITE CONSIDERATIONS

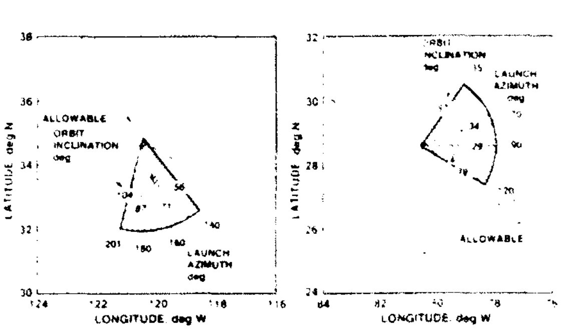
The orbit requirements for the GEO and LEO missions discussed earlier can be met by launching from the two main launch sites in the US, Cape Canaveral in Florida, and Vandenberg in California. Figure 13 shows the orbit inclinations that are achievable from each of the two sites. All targets at inclinations equal to or less than 57 deg can be reached from Cape Canaveral, while the higher inclination orbits , including polar and sun-synchronous, are accessible from Vandenberg. The Atlas or Delta boosters would launch the GEO satellites due east out of Cape Canaveral, while single LEO satellites could be launched on the small launch vehicles or ballistic missile-derivatives from either launch site. Large multi-satellite constellations, such as Iridium-like concepts could be launched by Delta or Atlas II out of Vandenberg. An Atlas II pad is being built at Vandenberg. For the ballistic missile-derivatives, the launch site at Cape Canaveral would have to be re-activated, and dedicated launch sites would need to be established at Vandenberg. The estimated cost for an above-ground ballistic missile-derivative space launcher pad is about $5M. The air-mobile Pegasus offers launch platform survivability and the maximum flexibility in launch site location and orbit inclination. An advantage of the Pegasus is that for target areas at latitudes lower than the 28 deg of Cape Canaveral, launching at inclinations equal to the target latitude provides an orbit that allows the satellite more consecutive passes over the target than the two times per day achievable if it were launched from Cape Canaveral. It should be noted that a cost penalty is incurred to exploit this advantage.

Table 4 lists the coordinates of most of the space launch sites that exist (or in some cases are planned or are inactive) in the world, and Figure 14 shows their distribution. The fluid global environment that is bound to exist in the future and the fact that tacsat mission will most likely be multinational in nature (and perhaps a UN responsibility) require the consideration of sharing launch facilities. The Italian site at San Marcos, off the coast of Kenya, the French site at Kourou, French Guiana, and the Brazilian site at Alcantara are interesting because they afford the direct launch inclination orbits for lower latitude targets. The relocatable Taurus would be a candidate for using non-US launch sites but, again, a cost-effectiveness analysis would need to be carried out. The U.S. Pegasus, the proposed CIS Space Clipper commercial orbital injection system, the UK/CIS Interim HOTOL and the German Sanger could also be launched from multiple launch sites.

10. CONCLUSIONS/RECOMMENDATIONS

While it is next to impossible to predict with certainty the environment of the future and what peacekeeping military capability will be demanded in that environment , there are some things that, in the long-term, appear inevitable:

- There will be a proliferation of high technology weapon systems around the world in the hands of nation-states that had not previously had access to such weapons.



Figue 13 US Primary Launch Sites

- Many of those systems and platforms will have the capability to deliver devices of mass destruction, i.e., chemical, biological, and nuclear devices.
- Many of the countries that have such weapons will have national goals that are in conflict with the interests of the U.S., its NATO allies, and other nations dedicated to world peace.
- Some of the countries who have such weapons will be dedicated to world terrorism and/or military attacks on their neighbors and/or internal civil strife destructive to the interests of world peace.

Table 4 Global Fixed Space Launch Sites

LAUNCH SITE	MAP DESIGNATION	COUNTRY	LATITUDE DEG MIN	LONGITUDE DEG MIN
EASTERN TEST RANGE	CAPE CANAVERAL	UNITED STATES	[illegible] 30 N	80 33 W
WESTERN TEST RANGE	VANDENBERG	UNITED STATES	34 36 N	120 36 W
WALLOPS ISLAND	WALLOPS	UNITED STATES	[illegible] 51 N	75 28 W
KAUAI TEST FACILITY	PALIMA POINT	U.S. (HAWAII)	[illegible] N	155 25 W
	KAHILI POINT	U.S. (HAWAII)	[illegible] 00 N	155 25 W
POKER FLATS	FAIRBANKS	U.S. (ALASKA)	65 07 N	148 28 W
KOUROU LAUNCH CENTER	KOUROU	EUROPE (ESA)	[illegible] N	52 46 W
SAN MARCO LAUNCH PLATFORM	SAN MARCO	ITALY	[illegible] 56 S	40 12 E
PLESETSK	PLESETSK	SOVIET UNION	62 48 N	40 24 E
KAPUSTIN YAR	KAPUSTIN YAR	SOVIET UNION	48 24 N	45 48 E
TYURATAM (BAIKONUR)	TYURATAM	SOVIET UNION	45 54 N	63 18 E
THUMBA EQUATORIAL STATION	THUMBA	(INDIA)	8 35 N	76 52 E
SRI HARIKOTA LAUNCH RANGE	MADRAS	INDIA	[illegible] 47 N	80 15 E
BALASORE LAUNCH STATION	BALASORE	INDIA	[illegible] 25 N	87 00 E
SHUANG-CH ENG-TZU LAUNCH COMPLEX	JIUQUAN	CHINA	40 25 N	99 50 E
XICHANG LAUNCH COMPLEX	XICHANG	CHINA	28 36 N	102 18 E
TAI YUAN LAUNCH COMPLEX	TAI YUAN	CHINA	[illegible] 46 N	112 30 E
WUZHAI LAUNCH COMPLEX	WUZHAI	CHINA	38 35 N	111 27 E
KAGOSHIMA SPACE CENTER	KAGOSHIMA	JAPAN	[illegible] N	131 05 E
OSAKI LAUNCH SITE	OSAKI	JAPAN	30 24 N	130 58 E
TAKESAKI LAUNCH SITE	TAKESAKI	JAPAN	30 23 N	130 58 E
WOOMERA LAUNCH SITE	WOOMERA	AUSTRALIA	[illegible] 07 S	136 32 E
ISRAELI LAUNCH COMPLEX	YAVNE	ISRAEL	[illegible] N	34 27 E
ANDOYA ROCKET RANGE	ANDOYA	NORWAY	[illegible] N	6 0[illegible] E
	ESRANGE	SWEDEN	68 30 N	21 00 E
	ALCANTARA	BRAZIL	[illegible] S	44 23 W
	MAR CHIQUITA	ARGENTINA	[illegible] 30 S	57 30 S
	LA RIOJA	ARGENTINA	26 20 S	66 50 S
CAPE YORK SPACE PORT	CAPE YORK	AUSTRALIA	[illegible] 30 S	143 00 E
	BARREIRA DE INFERNO	BRAZIL	25 55 S	35 08 W
	PANEUNGPEUK	INDONESIA	TBD	TBD
ANBAR RESEARCH BASE	TBD	IRAQ	TBD	TBD
KIRUNA LAUNCH STATION	KIRUNA	GERMANY	TBD	TBD
HAMMAGUIR LAUNCH STATION	HAMMAGUIR	ALGERIA	TBD	TBD

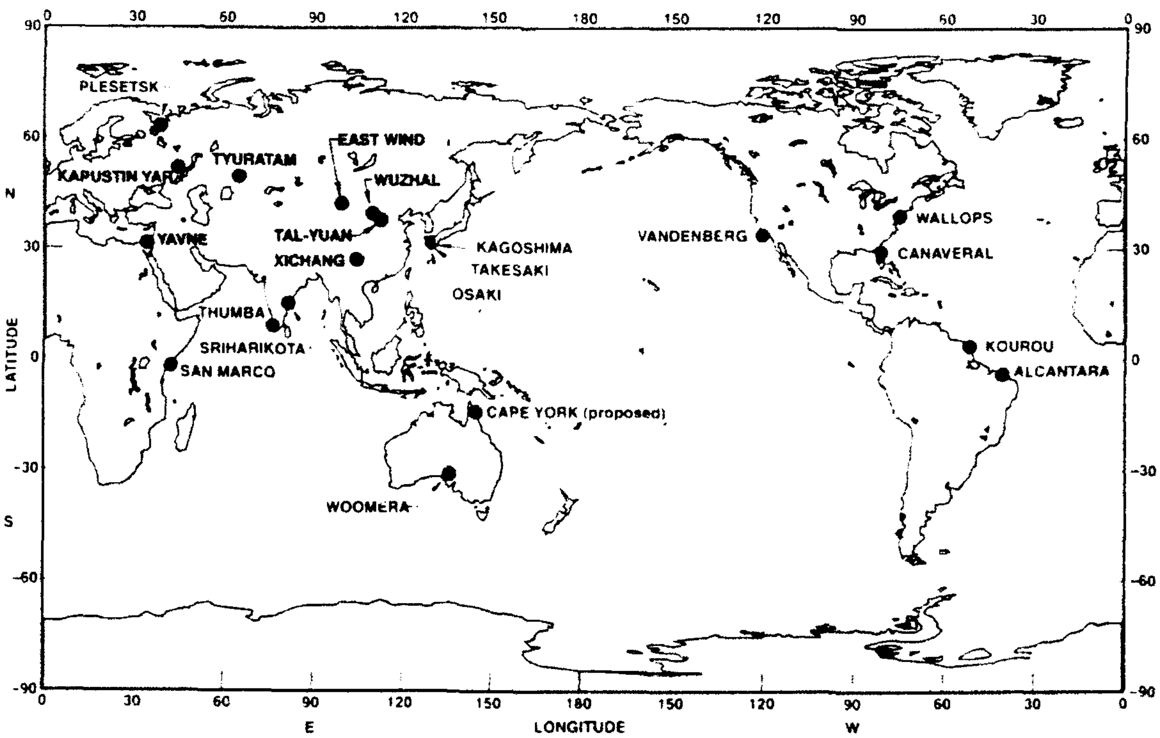


Figure 14 Distribution of Global Launch Sites

- Not only will such countries possess threatening military capabilities, they will also (eventually) possess the surveillance and reconnaissance capabilities to observe US and allied activities and be able to deliver their weapons at long range with precision.
- The response to unwarranted aggressive behavior, in order to be effective, will have to be very rapid multinational action, resulting in severe coordination problems in command, control, communication, intelligence-gathering and distribution, and treaty and arms control verification.

These changing environmental conditions, which are conjectured to evolve over the next thirty years, place increasingly demanding requirements on tactical orbital systems and hence on responsive launch services. The integrated need for launch services, ranging from small current launch vehicles, such as Scout, to advanced future launch systems, such as NASP, must be assessed in the light of the expected near-term and far-term tactical mission requirements. This paper characterizes the range of launch services that could satisfy tacsat needs and documents the best available information on the corresponding launch systems. It is concluded that the tacsat missions identified herein can be adequately supported by US launch vehicles launched from the continental US. However, numerous other options are available and need careful study by the tacsat designer or planner. A bibliography is provided to identify additional source material on some of the launch systems.

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SPACECRAFT AND LAUNCH SYSTEMS FOR TACSAT APPLICATIONS

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1. SUMMARY

The ability of a tactical satellite (TACSAT) space system to fulfill its mission application with the desired capability, responsiveness, reliability and survivability, while at the same time achieving low cost objectives, is a tremendous challenge that can only be met if all of the system segments - launch, space and ground - contribute to meeting mission unique requirements. The emerging concepts for the development, deployment and operation of cost-effective TACSAT space systems are especially dependent on the flexibility and operability of their launch vehicle and spacecraft bus systems. Orbital Sciences Corporation (OSC) has privately developed two flexible yet cost-effective space launch vehicles - Pegasus® and Taurus™ - with significant and unique operational capabilities that enable TACSAT space systems to meet these challenges. The Defense Advanced Research Projects Agency (DARPA) has sponsored the first launch of both systems, with follow-on launches scheduled in support of U.S. Air Force, NASA, SDIO and commercial programs. In addition, OSC has developed a flexible, cost-effective, spacecraft bus - PegaStar™ - that makes common use of the Pegasus or Taurus final stage avionics and structure in an integrated systems approach, thereby optimizing the mass and volume available for payload sensors. PegaStar spacecraft for the Air Force and NASA are now in engineering and production.

2. INTRODUCTION

The world today is a different and changing place, characterized by dramatic and rapidly evolving new geopolitical, economic and national security structures and relationships. In response, the United States' and its allies' national military strategies and national security infrastructures that support them must now depart from the principles that have shaped them since the end of World War II. These strategies and infrastructures, which evolved to contain the spread of Communism and deter Soviet aggression, must now shift dramatically to focus on regional crises and wars.

The threats of this new world can be best characterized as unpredictable and complex - where will they occur, how fast must be the response, what will be the sophistication level of the weapons, who are the combatants, what are the political and military goals, etc? The United States' military force structure for employment against these threats will be characterized by fewer in number, more cost-effective systems, fewer forward-based elements, and much lower active-duty manpower.

The operational imperatives to be addressed in the employment of these forces will be characterized by: (1) "come as you are" conflict scenarios; (2) response to rapidly unfolding crises with a few critical engagements; (3) long-range application of power; (4) high levels of responsiveness and flexibility for all systems; (5) situational awareness on a non-linear battlefield; (6) maneuverability; (7) coupling of national and theater levels of command, control, and intelligence; and finally (8) joint international operations.

Some of our first insights into this new world of tomorrow's conflicts come from analysis of results of the Gulf War. One of the most dramatic was the importance of space systems to all aspects of the planning, execution, and success of this war. Current space systems, which had evolved in support of the cold war and strategic requirements, were called upon to provide direct support to tactical warfighters. Many high-level military and civilian leaders characterized the conflict as the first "space war" - a war in which space systems were absolutely essential to both the execution of the conflict and its success.

The capabilities of these current space systems to support a broad spectrum of tactical warfare mission areas was impressive considering their heritage, but in many cases, their lack of flexibility and responsiveness impeded their performance. If the six months prior to the conflict had not been available to significantly adjust and modify their space-based architectures and ground systems, the level of support would have been significantly degraded. Also, if the conflict had extended in time, some space assets would have probably needed immediate replacement - which would have been very difficult due to the responsiveness and availability of spacecraft and launch vehicles.

These and other lessons from the Gulf War on space systems support to the tactical user will be the stimulus for the entire national security community to re-evaluate the warfighting requirements for future space systems and the infrastructure needed to meet these requirements. The resulting future space systems will no doubt include TACSAT systems that will emphasize flexibility, responsiveness and cost-effectiveness in meeting the changing U.S. and Allied national security requirements. Many of the other papers in this symposium will probably address the architectures, requirements, sensor system, ground systems and employment strategies for these systems. Orbital Sciences Corporation has developed two launch vehicles and a spacecraft-bus that will provide flexible, responsive and cost-effective launch and on-orbit performance for future TACSAT systems.

3. PEGASUS

3.1 Introduction

The flight-proven Pegasus air-launched space booster (Figure 1) provides a cost-effective, reliable, and flexible means for deliv-

Presented at the AGARD Avionics Panel's Symposium on "TACSATS FOR SURVEILLANCE, VERIFICATION and C3I", Brussels, Belgium, 19 - 22 October, 1992.

ering satellites into low Earth orbit. The vehicle's first launch on 5 April 1990 placed a DARPA sponsored, National Aeronautics and Space Administration (NASA) Goddard Space Flight Center (GSFC) built 192 kg (423 lb) payload into a 505 x 685 km (273 x 370 nm) 94 degree inclination orbit. This launch validated the vehicle's unique air-launch concept, simple, robust design and horizontal integration methods. The second launch on 17 July 1991 delivered seven DARPA sponsored small experimental communication satellites, (Figure 2) weighing a total of 196 kg (431 lbs), into a 352 x 500 km (190 x 270 nm) 82 degree inclination orbit. Even though this orbit was lower than desird due to a staging anomaly, the second launch demonstrated the vehicle's robustness and capability to deploy multiple communications payloads.

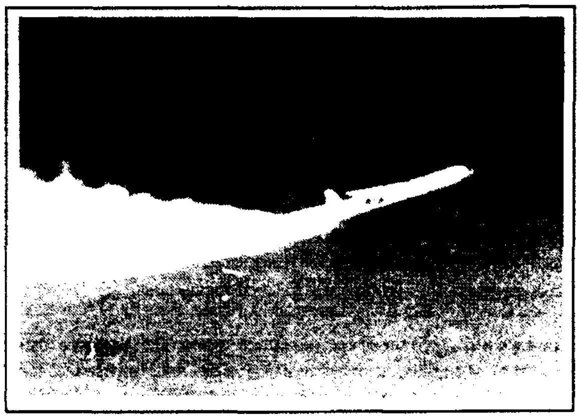

Figure 1. Pegasus in flight.

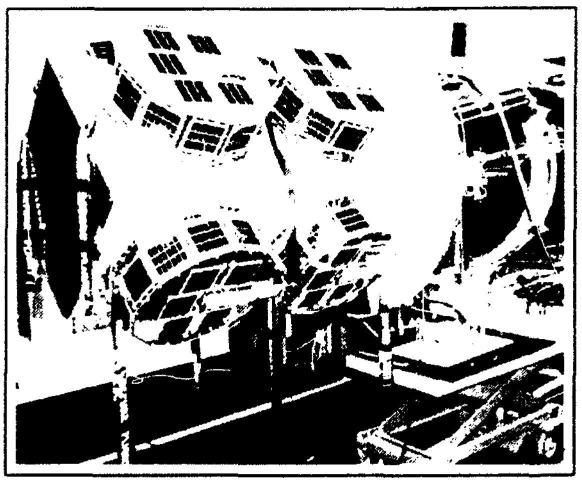

Figure 2. DARPA mirosats flown on F2 mission.

Pegasus, which holds a United States patent, is the product of a privately funded, three year joint venture of Orbital Sciences Corporation and Hercules Aerospace Company. The first two Pegasus missions were funded by DARPA as part of its Advanced Space Technology Program (ASTP) through the Advanced Systems Technology Offices (ASTO). Support was also received from the NASA Ames Dryden Flight Research Facility (DFRF) and the Air Force Space Division through agreements with DARPA. The vehicle's proven design and low cost have led to its selection as the U.S. Air Force Small Launch Vehicle (SLV), the Strategic Defense Initiative (SDIO) Experimental Satellite Expendable Launch Vehicle system (ESELVS) and for the NASA Small Expendable Launch Vehicle Services (SELVS) program. It has also been chosen to fly international and commercial missions. The vehicle is currently in production with ten payloads currently manifested for launch in the next eighteen months, and a targeted future average launch rate of approximately six vehicles per year. An increased performance version, Pegasus XL, is under development with a first flight scheduled to occur in late 1993.

Pegasus' unique air-launched approach provides unmatched operational flexibility. Pegasus is mated to the carrier aircraft and following pre-flight testing, the vehicle is carried to the launch point, which can be virtually any distance from the integration location. The carrier aircraft flies a pre-planned flight path to the designated launch point at which time the flight crew commands the launch of Pegasus.

3.2 Vehicle Description

The standard Pegasus vehicle (Figure 3) is 15.2 m (50 ft) long, has a diameter of 1.3 m (50 in), and weighs 19,000 kg (42,000 lbm). Pegasus XL is 17.4 m (57 ft) long and weighs 22,600 kg (49,800 lbm). Major components for both vehicles include:

- three graphite composite, solid-propellant rocket motors,
- a fixed, high-mounted graphite delta wing,
- an aluminum aft skirt assembly,
- three moveable graphite composite fins,
- a graphite composite avionics/payload support structure, and
- a two-piece graphite composite payload fairing.

Optional subsystems include:

- two payload separation systems, 59 cm and 98.5 cm (23.3 in and 38.8 in) in diameter,
- a restartable Hydrazine (N_2H_4) Auxiliary Propulsion System (HAPS), fourth stage and
- the PegaStar integrated spacecraft bus.

The vehicle's three solid rocket motors and payload fairing were developed for Pegasus by Hercules Aerospace. The 6.7 m (22 ft) span carbon composite delta wing provides lift during the early phases of flight. Three electro-mechanical actuated foam core graphite composite fins provide aerodynamic control throughout Stage 1 operation. Pitch and yaw control during Stage 2 and Stage 3 burn is provided by electro-mechanical thrust vector control (TVC) actuators. Roll control after Stage 1 separation, and three-axis control during coast phases and post orbital insertion maneuvers, is provided by 55 N (12.5 lbf) and 110 N (25 lbf) nitrogen cold gas thrusters located on the avionics subsystem. The graphite composite avionics structure and aluminum honeycomb deck support the payload and most vehicle avionics. A 1.3 m (50 in) outside diameter pyrotechnically separated two-piece graphite composite clamshell payload fairing encloses the payload, avionics subsystem, and Stage 3 motor. Two standard marmon clamp type payload separation systems (23 and 38 inch diameter) are available to support spacecraft deployment. The optional HAPS, fourth stage provides up to 73

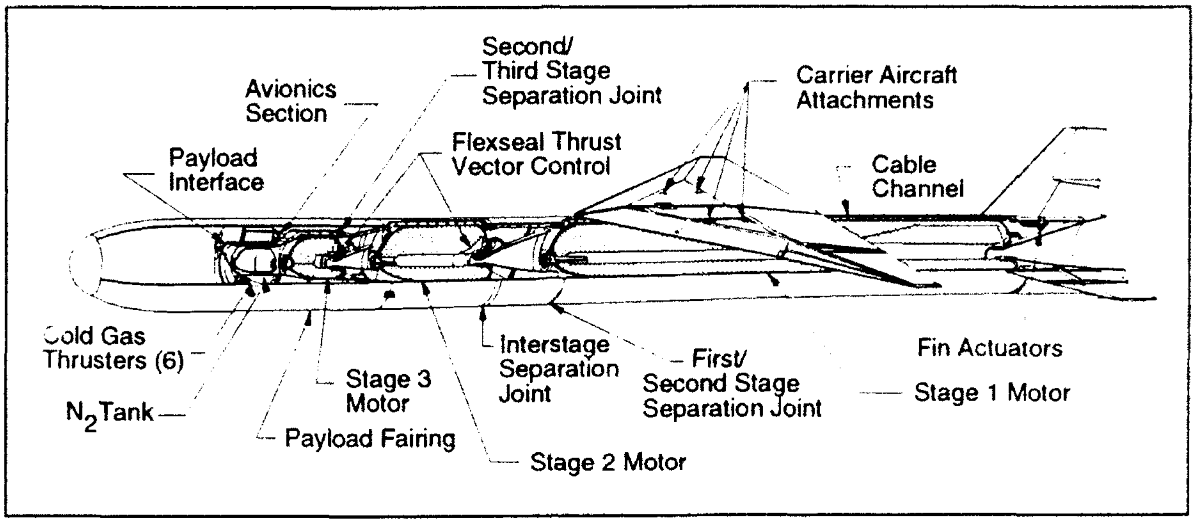

Figure 3. Pegasus cutaway.

kg (160 lbm) of N_2H_4 for orbit raising and/or adjustment. When combined with the Pegasus standard on-board Global Positioning System (GPS) receiver, HAPS provides autonomous precision orbit injection capability. The PegaStar integrated bus will be discussed later.

The Pegasus avionics system is simple, robust, and reliable. The vehicle's flight computer has two processors. A Motorola 68020 Central Processing Unite (CPU) commands all flight events and executes the vehicle's autopilot program. A Motorola 68000 CPU supports the autopilot processor and processes all vehicle telemetry. An inertial measurement unit (IMU) provides vehicle attitude, velocity and navigation information. All remote avionics units, which include Pyrotechnic Driver Unites (PDUs), Telemetry Multiplexors (MUXs), Thruster Driver Units (TDUs), and Thrust Vector/Fin Actuator Controllers, have integral microprocessors and communicate with the flight computer using digital RS-422 communication lines. All significant vehicle performance parameters are transmitted to the ground during flight using a single 56 kbps S-band telemetry channel. The six channel GPS receiver provides continuous position and velocity information to minimize the effects of IMU gyroscope drift and accelerometer bias errors.

3.3 Vehicle Integration

Final integration for Pegasus requires a minimum of ground support equipment (GSE) and facilities. Prior to being delivered to the field integration site, all components are integrated and tested to the highest possible levels. The vehicle is integrated horizontally (Figure 4), at a convenient working height which allows easy access for component installation, inspection, and test. Articulated transportation dollies eliminate all requirements for lifting motors in the field. Integration and test procedures enable vehicle components and subsystems to be thoroughly tested before and after final flight connections are made. Several "fly to orbit" simulations which exercise all actuators and pyrotechnic initiation outputs are conducted prior to launch.

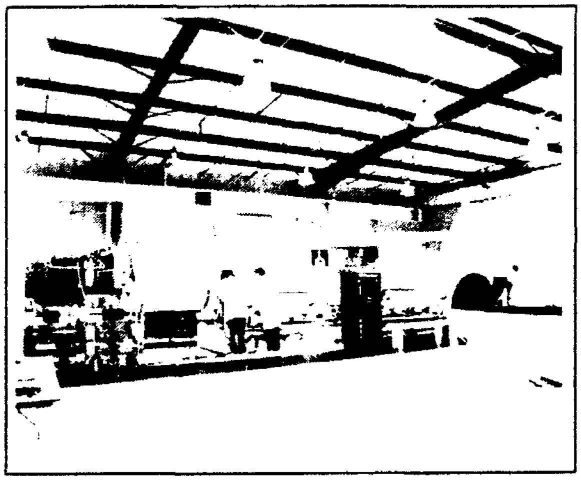

Figure 4. Horizontal vehicle and payload integration.

3.4 Launch Operations

Pegasus launch operations combine launch vehicle and aircraft operations, with emphasis on simplicity, flexibility, and operational discipline. The launch location flexibility inherent in Pegasus can significantly increase payload capability and orbital flexibility by eliminating many of the launch azimuth restrictions often imposed by fixed launch sites. After integration, mating to a B-52 or L-1011 carrier aircraft and pre-flight testing, the vehicle is carried to the launch point, which can be virtually any distance from the integration location. For launch, Pegasus is carried to a nominal level-flight drop condition of 12,200 m (40,000 ft) at high subsonic velocity. After release, the vehicle free falls to clear the carrier aircraft, while executing a pitch-up maneuver to achieve the proper attitude for motor ignition. After Stage 1 ignition, the vehicle follows an autonomously guided, lifting-ascent trajectory to orbit. Future missions will incorporate GPS into the tracking and guidance system of Pegasus resulting in an autonomous range capability.

3.5 Payload Capability and Interfaces

Pegasus' payload capabilities to polar orbits (assuming a 36 degree launch latitude) are summarized in Figure 5. Information regarding payload performance to elliptical and other inclination orbits, payload services and payload environments can be found in the Pegasus Payload User's Guide. The standard payload fairing (Figure 6) can support payloads as large as 2.21 m (87 in) long and 1.1 m (44 in) static diameter. The fairing can be extended in length 15 cm (6 in) increments up to 60 cm (24 in) and access doors can be repositioned or added as optional services.

3.6 Pegasus Operational Flexibility for TACSAT Applications

The Pegasus air-launched space system is unique in its capabilities to enable truly flexible, responsive, cost-effective access to space. These enabling capabilities, when combined with a modular spacecraft bus such as PegaStar and evolving sensor concepts, offer dramatic new possibilities in meeting the evolving national security requirements of tomorrow's world. The Pegasus system can be integrated and operated from remote secure airfields anywhere in the world due to its simplicity and minimal facilities and manpower needed. Dispersal of several Pegasus systems over a wide geographic area could also provide survivability levels unachievable by current launch systems.

With pre-planned positioning of vehicles and facilities, a Pegasus launch could be called up in hours - not days - to support new operational needs or reconstitution of an on-orbit asset. Since no pad refurbishment is required, quick turnaround in hours - not months - is also possible. Using more than one carrier aircraft and integration facility would allow surges of multiple launches in a single day if required. All of these scenarios for operational employment could be accomplished by small military or contractor teams.

Pegasus' omni-inclination launch capability (especially using OSC's autonomous range concept) provides unique flexibility to meet mission requirements and eliminate doglegs, thus improving payload performance to orbit. This capability also provides for first pass coverage of any point on the earth. Also of key operational importance is the minimal impact of weather on Pegasus launch system availability. Other than typical aircraft flight limitations, flight to the drop point above most weather systems is possible with a flexibility to adjust to new drop points in real time. There are also no trajectory limitations due to upper air winds, thus the Pegasus launch system readiness rates far exceed ground launched systems.

The Pegasus launch system provides national security space system strategists and planners with unique capabilities which when combined with evolving small spacecraft bus and sensor technologies, can enable flexible, responsive, cost-effective and, in some cases, undreamed of support to the warfighters in meeting new and evolving national security requirements.

4. TAURUS

4.1 Introduction

The Taurus™ space booster has been developed to provide a capability to quickly and efficiently integrate a launch vehicle and payload for the rapid launch of small satellites from either a remote or fixed launch site. This ground transportable launch system is erectable within five days of arrival at a "dry pad" launch site and, after set-up, respond to a launch-on-demand requirement within 72 hours. Initial launch capability of the Taurus launch system will occur in the first half of 1993. The first flight of the Taurus Standard Small Launch Vehicle (SSLV) is

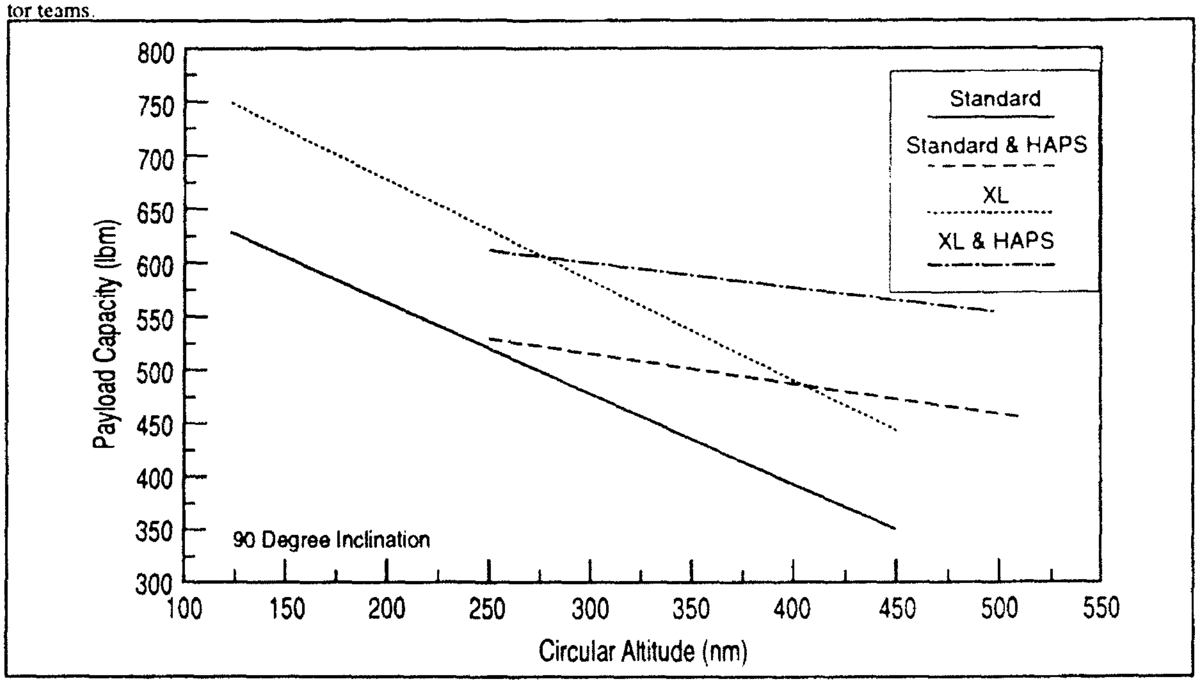

Figure 5. Pegasus performance options to 90° inclination.

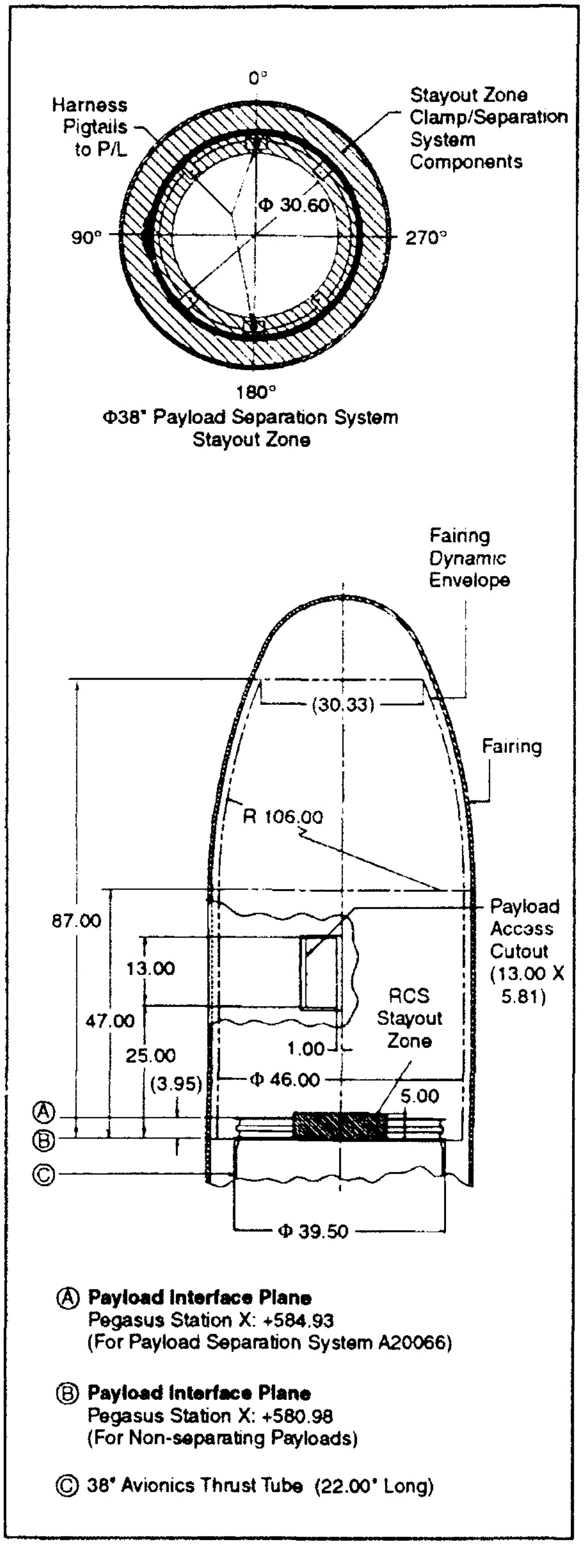

Figure 6. Standard and XL payload fairing dynamic envelope with 38" diameter payload interface.

sponsored by DARPA and will deliver a DoD payload to a low Earth orbit. The launch vehicle and system developed to satisfy these DoD requirements has also been designed to respond to commercial launch service applications requiring better response to meet market opportunities and with less bottom line impact. More capable Taurus configurations are being developed for introduction in 1994.

OSC has endeavored to develop user-friendly electrical and mechanical payload interfaces which facilitate integration of a wide variety of payloads. The Taurus vehicle's simple, robust design ensures maximum reliability and significantly reduces launch site manpower testing and support infrastructure requirements. Horizontal integration methods greatly simplify vehicle assembly and payload integration.

The Taurus space booster launch system currently in development as the SSLV for DARPA is a four-stage solid-fuel design composed of the flight-proven Pegasus motor stack and avionics mounted on top of a Thiokol TU-903 (Peacekeeper) solid rocket motor. (A Thiokol Castor 120 motor, now in test, will be used on the commercial Taurus). The Taurus SSLV system is composed of a flight vehicle and ground support equipment designed for easy transportability and rapid setup and launch from an unimproved concrete pad or from a modest fixed gantry. All propulsion elements of the DARPA Taurus vehicle have been flight-proven on the Pegasus or Peacekeeper programs. The remaining Taurus subsystems are virtually identical to those successfully flown on the Pegasus vehicle.

The standard vehicle design, shown in Figure 7, incorporates six major elements: four solid fueled stages, a payload fairing and an avionics assembly. The Pegasus-derived motors retain their Stage 1, 2 and 3 designations from that program; the Thiokol Castor 120 is designated Stage 0. The vehicle is designed to be integrated and tested at a launch site integration facility after completion of all stage-level assembly and testing at the factory. This procedure is less complex and more cost-effective than the procedures required for larger ground-launched vehicles which proceed through a full integrated factory system test followed by disassembly into stages for shipment and subsequent reassembly and second integrated systems test at the launch pad. With final integration and testing only at the launch integration site, the Taurus approach minimizes the handling of the solid rocket motors while simultaneously streamlining the vehicle acceptance test process. Extensive factory testing by both OSC and its vedors ensures that all hardware is ready for flight once it reaches the pad.

A complete set of standard and optional services is available to support specific payload requirements including: inertial orientation prior to separation; direct down-link of critical payload telemetry during prelaunch operations and launch; electrical power for payload heaters and other payload components during prelaunch operations; payload fairing access doors and Radio Frequency (RF) windows; filtered, conditioned air and dry nitrogen for purge of the payload environment during launch integrations.

For launch, Taurus is transported to the launch site using a wide-

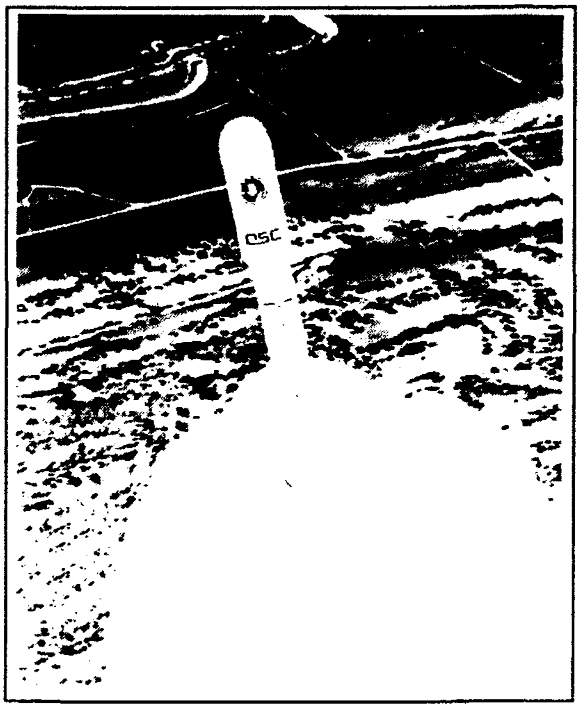

Figure 7. Artist's concept of Taurus lift-off.

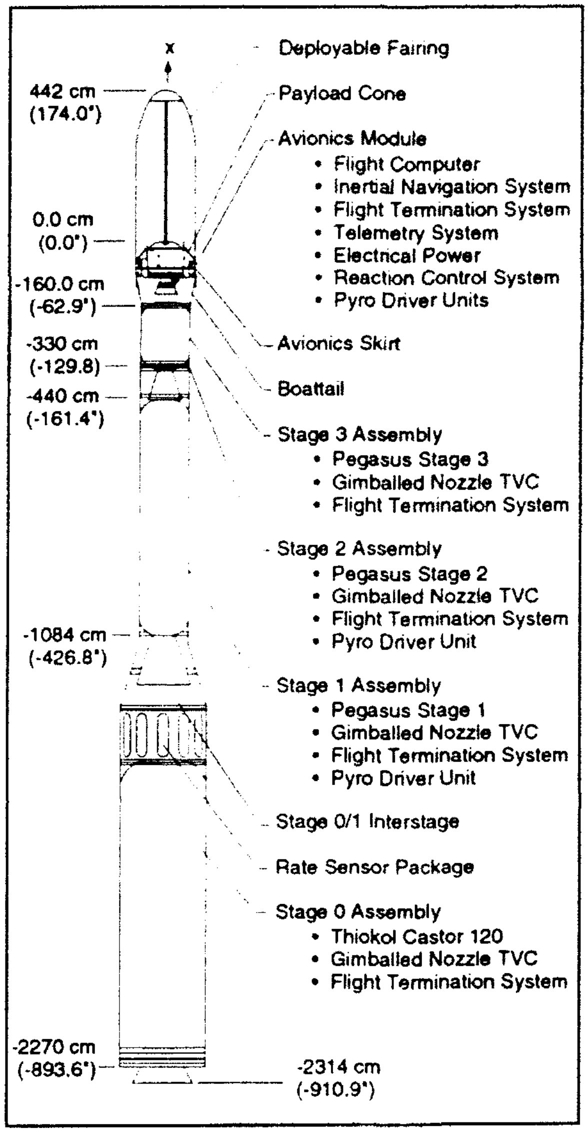

Figure 8. Taurus subsystems.

body flatbed truck for Stage 0 and an assembly integration trailer (AIT) for Stages 1, 2, 3 and the fairing with encapsulated payload. Stage 0 is received directly from the factory in a ready for flight configuration. Stages 1, 2 and 3 are received and tested at the launch site integration facility prior to arrival of the payload. Payload integration and testing usually requires no more than two weeks with Stage 3 mating, payload closeout and encapsulation occurring 96 hours prior to launch.

4.2 Vehicle Description

Major vehicle components as shown in Figure 8 include:

- four composite case solid-propellant rocket motors,
- 3-axis inertial attitude control,
- on-board global positioning system (GPS) receiver for navigation accuracy,
- graphite composite avionics/payload support structure, and
- two-piece honeycomb composite payload fairing with over 5.15 m^3 (175 ft^3) available for satellite payloads.

Option subsystems include:

- 98.5 m (38.8 in) payload separation system,
- restartable Hydrazine (N_2H_4) Auxiliary Propulsion System (HAPS), and
- PegaStar integrated spacecraft bus.

The Taurus vehicle's four solid rocket motors and payload fairing use flight-proven technology derived from Pegasus and other operational systems to provide an innovative, cost-effective approach for delivering a spacecraft to orbit. Pitch and yaw control during Stage 0 operation is provided by a hot gas driven turbohydraulic thrust vector actuation (TVA) system. Pitch and yaw control during Stage 1 operation is accomplished with a cold gas blowdown hydraulic TVA, while Stage 2 and Stage 3 pitch and yaw control is provided by electro-mechanical thrust vector control (TVC) actuators. Roll control after Stage 1 separation is provided by 55N (12.5 lb) and 110N (25lb) nitrogen cold gas thrusters located on the avionics subsystem. The graphite composite avionics structure and aluminum honeycomb deck support the payload and most avionics. A 1.6 m (63 in) outside diameter pyrotechnically-separated, two-piece graphite composite payload fairing encloses the payload. Fairing halves are retained by a base aluminum attach ring, two externally mounted clampbands and an internal mounted forward attach bolt. Separation is accomplished via a marmon clamp at the base aluminum ring and redundant bolt cutters at the clamp assemblies and forward attach bolt. Fairing hinges and hot gas actuators ensure positive controlled fairing separation. An optional 38 in Marmon clamp payload separation system is available. The optional HAPS provides up to 73 kg (160 lb) of N_2H_4 for orbit raising and adjustment. When combined with the Taurus standard on-board Global Positioning System (GPS) receiver, HAPS provides autonomous precision orbit injection capability. The optional PegaStar integrated spacecraft bus will be discussed later.

The Taurus avionics system is simple, robust, and reliable. All of the hardware components are identical to those used on Pegasus, with the exception of the addition of a reate gyro package to the base of Stage 1 and harness modifications to accommodate Stage 0. The software programs are identical except for the addition of Stage 0 function and mission peculiar modifications.

4.3 Vehicle Integration

Taurus field integration is straightforward and requires a minimum of launch support equipment (LSE) and facilities. All of the LSE is transportable and capable of setup at remote austere sites. The vehicle is integrated horizontally at a convenient working height which allows easy access for component installation, test and inspection. The use of standard serial RS-422 communication protocols throughout simplifies vehicle wiring, streamlines avionics testing and integration and significantly reduces test and custom LSE requirements. The integration and test process ensures that all vehicles components and subsystems are thoroughly tested before and after final flight connections are made. Several "fly to orbit" simulations exercise all actuators and pyrotechnic initiation outputs. Taurus integration activities are controlled by a comprehensive set of Work Packages (WPs) and Procedural Guides (PGs), which describe and document in detail every aspect of integrating the vehicle and its payload.

4.4 Launch Operations

Taurus launches can be conducted from either of the Department of Defense's Eastern or Western Ranges (ER/WR) as well as from the National Aeronautics and Space Administration (NASA) Goddard Space Flight Center's GSFC) Wallops Flight Facility (WFF) Range in Virginia. The first Taurus launch for DARPA is scheduled for the first half of 1993 from Vandenburg AFB onto the Western Range using only a simple concrete pad to set up OSC's transportable launch support equipment. OSC's commercial Taurus launch service will be conducted utilizing a modest fixed gantry with a rotating service structure. OSC's primary commercial Taurus launch facilities will be established at launch sites optimum for the particular mission being supported -- Vandenburg AFB, CA; Wallops Island, VA and Cape Canaveral are likely locations.

A Taurus payload interface checkout facility is available as a standard service at OSC's Fairfax, VA engineering facility for initial payload to launch vehicle functional test and checkout. Housing an engineering test model of the Taurus avionics assembly and a complete set of avionics and payload test equipment, this Vehicle Systems Integration LAB (VSIL) will be used for the development of mission peculiar launch vehicle software, qualification of mission peculiar avionics services and validation of payload integration procedures and flight checklists. The facility will also be equipped to conduct launch vehicle mission simulations with payload hardware and software in the loop.

4.5 Payload Capability and Interfaces

Payload capability of the DARPA Taurus vehicle to some typical orbits is shown in Figure 9. The Taurus XL configuration uses the Pegasus XL stretched motors and the Castor 120 whereas the Taurus XL/S adds two strap-on rocket motors to Stage 0. Figure 10 shows performance to a 90° orbit from Vandenberg. Specific information regarding payload performance to other orbits can be determined by contacting OSC's Office of Business Development. The standard payload fairing can support payloads as large as 2.82m (110 in) long and up to 1.38m (54 in) in diameter (Figure 11). Funded studies are in progress to address increasing the fairing diameter.

Orbit	DARPA Taurus Capability
250 nm, 28°, (Orion 38 Stg 3)	2450 lbs (1114 kg)
Geosynchronous Transfer (STAR 37 PKM)	860 lbs (391 kg)

Figure 9. DARPA Taurus capability.

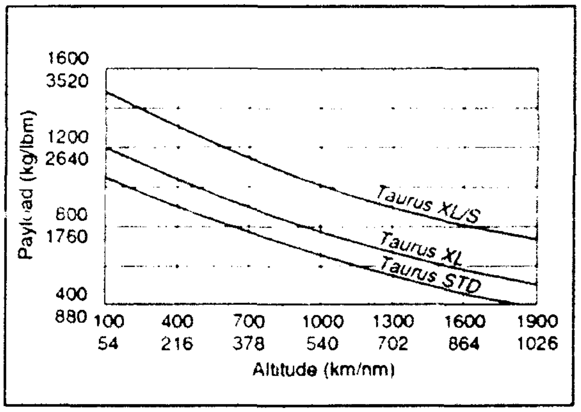

Figure 10. Taurus configurations' performance to 90° inclination.

The Taurus vehicle offers a number of standard services. The standard 1.6m (63 in) diameter Taurus payload fairing offers over 5.15m³ (175 ft³) for payload use. The fairing is designed to encapsulate the payload in a payload integration facility following checkout. A class 10,000 environment can be maintained inside this encapsulated cargo element (ECE) at all times from encapsulation through ECE transport, vehicle integration and up until launch. As a standard service, OSC provides one 12-inch square access door in each half of the fairing, located to the requirements of the payload. These doors provide all the access, including late access on the gantry, to the payload once encapsulation is completed. As an option, OSC can reposition the access doors, provide larger or additional access doors or RF-transparent panels. The payload mechanically mounts to the standard Taurus separation system via 60 fasteners about a 38.81-inch diameter bolt circle.

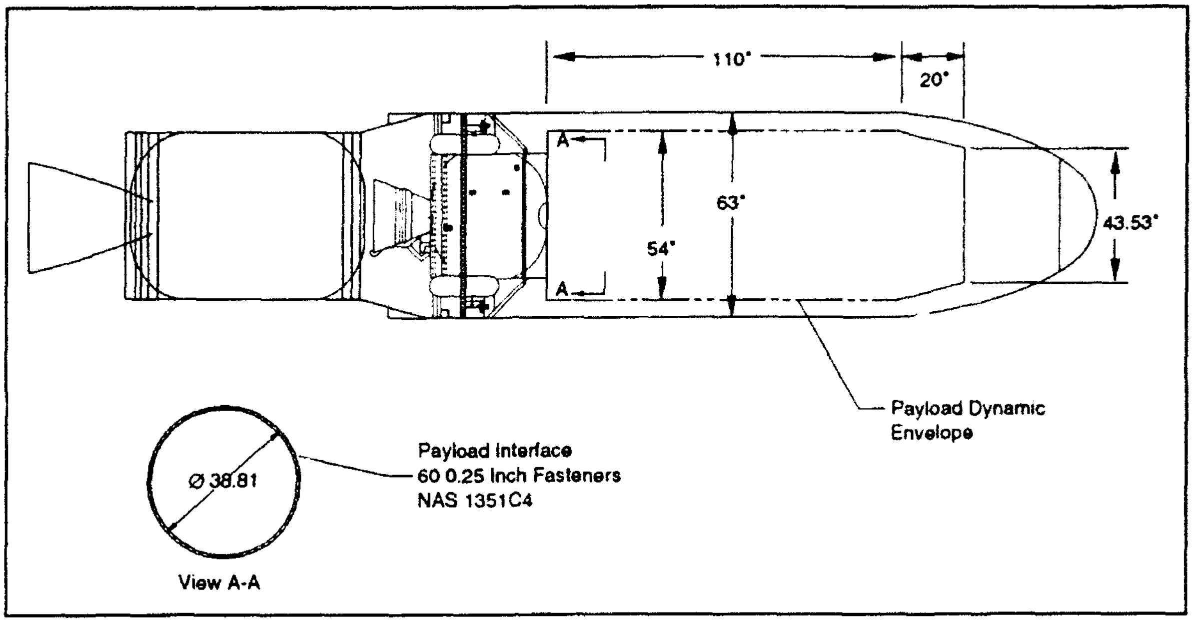

Figure 11. Taurus payload fairing dimensions.

The Taurus payload electrical interfaces are established through three connectors - power and signal, pyro, and RF. The power and signal connector provides six payload passthroughs (prior to launch), eight payload discrete commands, four payload discrete talkbacks and a payload separation sensing breakwire. The power and signal connector also supports an optional payload RS-422 telemetry stream. The pyro connector can initiate up to five redundant payload pyrotechnic events. In addition, the RF connector supports payload RF transmission via the existing Taurus antenna system. Detailed information on payload services and payload environments provided by Taurus can be found in the Taurus Users Guide available on request from OSC.

4.6 Taurus Operational Flexibility for TACSAT Applications

The Taurus launch system, has attributes for flexible, responsive and cost-effective launch far beyond existing launch systems. With its ground mobility and compatibility with air transport, Taurus can provide significant levels of survivability through dispersal. This is greatly enhanced by the fact that Taurus can be launched from an "unimproved, dry pad" - i.e., the end of a runway. The Taurus system has been designed and will demonstrate meeting DARPA's requirements for a launch site and vehicle set-up of less than five days and launch-on-demand within 72 hours after set-up. The vehicle can be maintained in this 72-hour launch ready condition for months if required to meet system operational requirements. Turnaround time to erect and launch another vehicle is measured in days, not months, providing significant surge capability. Finally, the Taurus fairing encapsulation system allows for flexible pre-launch payload integration and the capability to store the encapsulated payload for long periods awaiting the launch.

All of the above capabilities can support LEO as well as GEO missions. These Taurus launch system unique capabilities can be exploited to develop and support innovative, flexible, responsive and cost-effective space systems to meet the new world requirements of future space systems.

5. PEGASTAR™

5.1 Introduction

In 1989, OSC began work on several low-cost, multi-purpose spacecraft projects. One of these, the PegaStar integrated spacecraft bus is designed for use with Pegasus and Pegasus-derived launch vehicles, including Taurus. PegaStar is a key element of the company's ability to offer comprehensive space launch and satellite services. Using many of the same systems that operate the Pegasus vehicle, PegaStar is being built around the third stage of Pegasus (minus the motor which is separated after attaining orbit) to provide the "house-keeping" services necessary to support OSC or customer-provided instruments, communications devices and other sensors on-orbit. In early 1991, OSC entered into a contract with the U.S. Air Force Space Test Program for the production of a sun-pointing PegaStar spacecraft to support several Air Force scientific experiments, as well as an environmental monitoring data contract with NASA Goddard for the provision of ocean color data using an instrument mounted on a nadir pointing PegaStar spacecraft.

5.2 Vehicle Description

OSC's PegaStar integrated satellite (Figure 12) approach provides for a more efficient use of the volume and mass available from a Pegasus or Taurus launch compared to a separable satellite design. By eliminating the weight and cost necessary to provide separate avionics, control, structural and data systems on both the payload and launch vehicle, PegaStar will enable a user to place heavier, payloads into orbit for the same or lower launch cost than would be otherwise required. This performance advantage (typically 15% to 20%), makes one or more design enhancements possible: additional payload mass, additional system redundancy, less complex system packaging, reduced fuel requirements, increased margin of reserves on launch, or a combination of these benefits. The PegaStar concept increases packaging efficiency, thereby reducing cost and improving the throw weight capacity of the Pegasus or Taurus launch system. A typical PegaStar spacecraft dry mass is 173 kg (382 lbm), with a payload capacity of up to 143 kg (315 lb). PegaStar's modular design and flexible avionics architecture enable the support of a wide variety of sensor and mission applications. Our first two PegaStar spacecraft, the USAF APEX mission and the NASA SeaStar

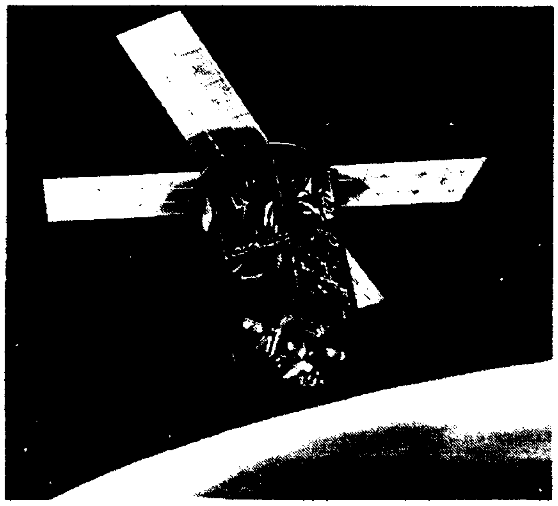

Figure 12. Nadir pointing PegaStar spacecraft.

mission, are scheduled for flight in 1993. A summary of each of these mission spacecraft is shown in Figure 13.

Spacecraft	APEX	SeaStar
Launch Year	1993	1993
Mass	247 kg	245 kg
Power Peak	375w	500w
Attitude Control	3 Axis, 0.5°	3 Axis, 0.1° 1.23 mrad Knowledge
Transmitters	S Band	2 S Band 2 L Band
Propulsion	None	N_2H_4
Orbit Determination	GPS	Autonomous GPS to w/in 100m
Lifetime	1 Year Design 3 Year Goal	5 Year Design, 10 Year Goal
Notes	DoD HDBK 343 Class C 64 mBit Data Recorder	Fully Redundant 1.25 GBit Data Recorder 2.6 Mbps Down-link
	In Test	In Fabrication

Figure 13. PegaStar spacecraft capabilities.

5.3 PegaStar Operational Flexibility for TACSAT Applications

The modular design flexibility of the PegaStar spacecraft bus enables the integration of sensor systems with a broad spectrum of system capabilities. Structural design flexibility allows for the incorporation of the sensor and its systems onto separate, modifiable panels which can be integrated independently but in parallel with the spacecraft subsystem. The distributed microprocessor bus communication system enables the straightforward integration of sensor peculiar systems as well as the reconfiguration of the basic PegaStar spacecraft attitude control and TT&C systems to meet the unique sensors requirements. Also modular in design approach is the spacecraft power system, which can be configured with fixed arrays, sun-tracking arrays, body arrays, choice of silicon cells or gallium-arsinide cells and electrical power supply configurations to support power levels from 100 watts to 1000 watts.

The design flexibility of the PegaStar accommodates a wide range of sensor suites to enable new and innovative approaches to the total system design of future space systems. Working closely with the sensor designers to develop standard and compatible interfaces, these new approaches to sensor/spacecraft integration could have dramatic payoffs in the development and deployment of flexible, responsive and cost-effective space systems to provid support to the warfighter.

6. APPLICATIONS

TACSAT systems can provide important benefits in military engagements ranging from covert operations to regional and global battles. Three examples are now provided for the use of flight-ready TACSAT technologies in a relatively inexpensive tactical surveillance system, a commercial TACSAT communication network and a complementary TACSAT fleet that would work in concert with existing military satellites to collect important theater meteorology data.

6.1 Surveillance

Tactical surveillance satellites intended primarily to support theater commanders have been discussed for at least the last five years. From these discussions, it has emerged that such TACSATs should have the following operational characteristics:

- Improved temporal resolution compared to current systems, that is short revisit times over areas of interest made possible by a distributed satellite constellation;
- Adequate spatial resolution to be operationally useful, which generally translates into a ground-separation distance scale of a few meters;
- Targeting authority delegated to the ultimate users; extensive on-board processing of surveillance data; and processed data distributed directly to such users, often with real-time and/or simultaneous downlinks to several locations;
- Peacetime daily availability for training exercises and other routine purposes;
- Short satellite lifetimes (3 to 5 years) so that new capabilities and technologies can be rapidly introduced; and
- Relatively low cost to build, launch, operate and permit rapid establishment/augmentation of constellations in times of crisis or war.

Current technology makes it possible to build passive optical surveillance satellites weighing around 363 kg (800 lb) and active synthetic aperture radar satellites weighing 680 kg (1,500 lb) for $20-30 million per satellite. With the advent of Pegasus and Taurus, it is already possible to launch such satellites for $10-20 million and to do so with the necessary degrees of responsiveness in initial launch and replenishment.

This is not to imply that a single TACSAT network will solve all the warfighter's intelligence and surveillance needs. Operational commanders will continue to rely on support from larger space systems, as well as on an array of battlefield surveillance systems. Nevertheless, there is assuredly great value in such a tactical surveillance system in the new defense environment of the 1990's. For instance, as was dramatically reinforced in the Gulf War, the operational cycle of target selection, planning, execution and data delivery was about three days with existing surveillance satellites. Substantially improved time responsiveness of point and area space surveillance to operational commanders' needs would be of very high utility in a crisis period.

6.2 Communications

By the mid-1990's commercial, TACSAT like, data communications networks will be operational, followed later in the decade by voice networks. Already, a TACSAT voice and data telecommunications system has been demonstrated for worldwide applications. Today, OSC's Orbital Communications Corporation subsidiary is building a constellation of 26 small satellites to provide omnipresent two-way message and data communications. ORBCOMM™, as it's commonly known, will provide high services availability virtually every place on Earth at low, consumer-level equipment and service costs. Its two-way packet data communications services will be available to military users without the requirement for Government funded development of a unique communications system.

Several promising military applications of ORBCOMM are being discussed within the services and DoD. Desert Storm highlighted a need for deployable field communications for the rapidly mobile forces. Field logistics personnel needed a better way to place material orders and to receive confirmation of status. It is reported that millions of tons of material were doubled ordered and shipped to Saudi Arabia because of the absence of communications capability that could operate easily over a very wide area. Tracking of shipments provided for a still larger logistic problem with supplies sometimes in theatre, but their location was unknown to the user. ORBCOMM's two-way, pocket-portable, operate-anyplace system will provide the missing link in the supply chain.

In addition, GPS showed its tremendous operational value in Desert Storm. For the first time, field units knew very precisely their location in an environment with few geographic reference points. Now, with ORBCOMM, we are ready to go a step farther by allowing those field units to easily transmit their locations and unit numbers plus other critical data back to the commanders at any organizational level desired. We are working with manufacturers of GPS receivers to build in the capability to GPS equipment for the command structure to request automatic transmission of the location of platoons or mechanized equipment and for the field units to report their locations and other vital information on demand. The marriage of GPS and ORBCOMM capabilities may also address, in part, the problems of friendly fire casualties and significantly enhance the search precision and probability of success of downed pilot recovery.

6.3 Meteorology

Finally existing military satellites could be supplemented by a network of TACSAT systems to enhance our knowledge of the environment. Efforts to clear up the "fog of war" have been shown to be crucial to battle success for centuries. For example, one constellation of 24 TACSAT GPS receiver platforms could provide several tens of thousands of randomly-distributed occultation measurements per day of the properties of the atmosphere from the troposphere to the ionosphere. These measurements would provide research data on greenhouse gases and produce "weather maps" of the ionosphere to better understand the link between the sun and our climate. Alternatively, small imaging payloads in the visible regime could provide an overhead look at target areas providing a pre-flight estimation of cloud cover prior to committing a fleet of bombers to a mission. Such payloads would not require the resolution required of surveillance systems.

7. CONCLUSIONS

This paper provides a description of several innovative launch vehicle and spacecraft systems and offers brief examples of their flexibility and capability to cost-effectively meet the unique warfighting requirements that must be addressed by evolving TACSAT systems. The timing and extent to which these systems will become integrated into the infrastructure of military forces is driven by a number of factors.

The euphoria which exists in the governments of Western nations, brought on by the breakup of the Soviet Union, will likely continue to have an effect on the defense spending of these countries for several years. Coupled with sluggish economic conditions, the U.S. is curtailing military expenditures at unprecedented rates. From a "good news/bad news" perspective, the bad news is this will likely reduce significantly the funding available for any new system development and production. But the good news is TACSAT systems were conceived in part to meet the challenge of declining military budgets and therefore do not require significant investment to bring them to fruition. It is in fact arguable that these TACSAT systems might well flourish in difficult economic times.

Second, the widely distributed threat environment of today and tomorrow strengthens the case for systems which are responsive to new tactical and strategic realities. TACSATs provide not only the focus needed for regional conflict and monitoring, but the low-Earth orbits common to such systems permit an orbit-by-orbit reallocation of assets to account for ever changing priorities. The pace of technology is such that both the capability and capacity to address worldwide hot-spots is available from responsive space systems like PegaStar, Pegasus and Taurus.

Finally, as with all new warfighting capabilities, the introduction of novel approaches to meeting the requirements of the battlefield have often met with some resistance - such has been the case with TACSATs. Now that a few systems have been flown and more importantly now that the commercial market is finding applications for such systems, the introduction of TACSATs into the national security system architecture will rapidly expand. Whereas prior to the Gulf War such TACSATs could be ignored as unproven, now there is documented evidence of the value of these systems and the users are beginning to identify additional applications where an augmentation in capability, or an entirely new function, can be fulfilled with a TACSAT. Just as the battle tank and the airplane needed to be proven, so too must the TACSAT earn its place on the battlefield. Having earned that place, TACSATs will become essential components of future military infrastructures and strategies.

Discussion

Question: OSC said that the corporate goal is to penetrate the commercial market. Of the 76 PEGASUS orders, how many are governmental and how many are free market?

Reply: Until now the priority is governmental (85%), but in the future OSC believes it can increase the commercial percentage.

OPTIMISATION DE LANCEMENTS MULTIPLES

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RESUME

Puisque les futurs lanceurs permettront l'emport d'une charge utile de plus en plus lourde, il est possible d'imaginer des lancements de plusieurs petits satellites sur des orbites différentes. Cependant, pour diminuer autant que possible le coût des manoeuvres de transfert de ces satellites, il est essentiel de déterminer l'orbite d'injection optimale qui minimise par exemple la masse d'ergols nécessaire à ces transferts.

La méthode utilisée pour résoudre ce problème d'optimisation complexe est l'algorithme du gradient projeté généralisé. Il permet de trouver l'orbite d'injection optimale et la trajectoire ascensionnelle correspondante qui minimisent le coût tout en respectant les contraintes imposées à la trajectoire de montée et aux transferts.

Deux exemples d'application seront exposés. Le premier est l'optimisation d'un lancement double vers deux orbites d'inclinaisons différentes, le second concerne la mise à poste d'une constellation positionnée sur des orbites de noeuds ascendants différents.

ABSTRACT

Since future launchers deliver more and more payload mass, it is possible to imagine multiple launches of small satellites into noticeable different orbits. Yet, so as to decrease possible high cost of satellite transfers, it is essential to determine the optimal injection orbit that minimizes for instance the ergol mass necessary to these transfers.

The method used to solve this complex optimization problem is the generalized gradient algorithm. It allows to find the optimal injection orbit that satisfies the miscellaneous constaints applied to the launcher and its trajectory, and tha minimizes the cost function. At the same time control laws and parameters of ascent phase and transfers are optimized.

This method application will be shown on two examples, the first concerning a dual launch on two differently inclined orbits; the other one concerns a constellation of different ascending node positions.

1 INTRODUCTION

Puisque les futurs lanceurs - comme Ariane V - auront la possibilité d'emporter de plus en plus de charge utile, on peut imaginer des lancements multiples de petits satellites sur des orbites différentes. En particulier, le lancement simultané de plusieurs satellites d'une constellation de radiolocalisation, ou de deux satellites aux missions différentes sur leurs orbites respectives, peut constituer une solution intéressante malgré le coût probablement important des manoeuvres de transferts après l'injection.

En conséquence, il est essentiel de déterminer l'orbite d'injection qui minimise les masses d'ergols ou les incréments de vitesses nécessaires aux transferts des satellites sur leurs orbites.

La méthode utilisée pour résoudre ce problème complexe d'optimisation fonctionnelle et paramétrique est l'algorithme du gradient projeté généralisé [1]. Cette méthode, développée à l'ONERA, permet de favoriser la satisfaction des contraintes lors des itérations, et donc de trouver une trajectoire optimale les respectant même si la trajectoire initiale en était loin. Ces contraintes peuvent être courantes (flux thermique maximal), intermédiaires (retombée d'un étage) ou finales (paramètres orbitaux), d'égalité aussi bien que d'inégalité. En outre, une fonction coût est minimisée comme la masse d'ergols ou les incréments de vitesses nécessaires aux manoeuvres de transferts des satellites. Les lois de contrôle du lanceur ainsi que des paramètres portant sur la trajectoire de montée et sur les manoeuvres des satellites sont optimisés. Ces manoeuvres peuvent être mono ou bi-impulsionnelles et elles induisent leurs propres contraintes.

La première partie concerne la représentation utilisée pour ce problème physique: modélisation du mouvement du lanceur et des satellites, lois de contrôle et paramètres optimisés, coût à minimiser et contraintes à satisfaire. La deuxième partie illustre les potentialités de l'outil d'optimisation: lancement double vers des orbites de même altitude mais d'inclinaisons différentes; mise à poste d'une constellation dont les orbites ont des noeuds ascendants différents. Dans chaque cas, l'intérêt de l'optimisation de l'orbite d'injection sera démontré.

2 POSITION DU PROBLEME

2.1 Mouvement du lanceur et des satellites

2.1.1 Modélisation du mouvement du lanceur

Si $\vec{R}$ et $\vec{V}$ représentent la position et la vitesse par rapport à un repère inertiel, les équations du mouvement du centre de gravité sont:

$$\dot{\vec{R}} = \vec{V}$$

$$\dot{\vec{V}} = \vec{g} + \frac{\vec{F_a} + \vec{F_p}}{m}$$

$$\dot{m} = -q$$

où m est la masse instantanée du lanceur.

$\vec{g}$ est l'accélération gravitationnelle (avec prise en compte de la perturbation due au J_2)

$\vec{F_a}$ et $\vec{F_p}$ sont les forces aérodynamique et propulsive

q est le débit instantané d'ergols.

Le vecteur d'état X(t) est de la forme $X(t) = (R, V, m)^T$.

Les lois de commande choisies pour déterminer l'orientation du lanceur sont l'assiette θ et l'azimut ψ, exprimés dans un repère inertiel. On suppose que le lanceur vole sans dérapage, l'angle de roulis ϕ est donc fonction de θ et ψ. Il est clair que la connaissance de l'orientation du lanceur permet la détermination complète de la direction et du module des forces aérodynamique et propulsive. En ce qui concerne la phase atmosphérique, le vol s'effectue à azimut inertiel constant et à incidence quasi-nulle après le basculement du lanceur. Le mouvement du lanceur dans l'atmosphère est donc déterminé par trois paramètres: azimut inertiel, vitesse de basculement et durée du basculement.

Le système différentiel précédent peut donc se mettre sous la forme suivante:

$$\dot{X}(t) = f(X, U, A, t)$$

où U est le vecteur de commande et A le vecteur constant des paramètres optimisables.

2.1.2 Modélisation du mouvement des satellites

Le mouvement des satellites est entièrement déterminé par les six paramètres orbitaux suivants: le demi-grand axe noté a, l'excentricité e, l'inclinaison i, l'argument du noeud ascendant Ω, l'argument du périgée ω et l'anomalie vraie v. Ces paramètres obéissent aux lois de Kepler.

2.2 Chronologie

Le vol du lanceur est émaillé d'une série d'événements particuliers comme le largage des étages vides ou de la coiffe, introduisant des discontinuités dans la masse et les forces propulsive et aérodynamique. De plus, la chronologie de ces événements est a priori inconnue. Pour traiter ces discontinuités, il est nécessaire de considérer des sections de trajectoires. Le passage d'une section à l'autre est déterminé par le changement de signe d'un certain critère, différent suivant les événements.

2.3 Fonction coût

Etant donné que les caractéristiques du lanceur, les conditions initiales et les éléments orbitaux finaux sont connus, si l'on néglige le temps de réponse du système de contrôle d'attitude du lanceur, le problème consiste à déterminer les lois d'assiette et d'azimut optimales procurant le coût minimal tout en respectant les contraintes.

2.3.1 Expression du coût

Le coût à minimiser peut être l'incrément total de vitesses à fournir pour permettre les transferts:

$$V_o = \sum_i \Delta V_i$$

où ΔV_i est l'impulsion du $i^{ème}$ satellite.

On peut aussi choisir de minimiser l'impulsion maximale des moteurs d'apogée de chaque satellite:

$$V_o = \underset{i}{Max} \mid \Delta V_i \mid$$

Dans ce cas, le programme tend à égaler les différentes impulsions de transfert.

La fonction peut aussi concerner les masses d'ergols nécessaires pour effectuer les transferts d'orbites. La masse d'ergols requise pour la $i^{ème}$ manoeuvre est:

$$M_{ergi} = m_i \left(\exp \left(\frac{\Delta V_i}{g_0 Isp_i} \right) - 1 \right)$$

où m_i est la masse finale du $i^{ème}$ satellite.
Isp_i son impulsion spécifique, exprimée en secondes.

Les caractéristiques du moteur d'apogée correspondant, capable de fournir cette impulsion grâce à sa masse d'ergols, seront celles de la gamme de moteurs MAGE (Isp de 310 s). En outre, afin de simplifier le calcul de la masse sèche du moteur, on considérera un coefficient de structure constant de 10% (rapport de la masse sèche sur la masse d'ergols).

Comme précédemment, le coût peut être exprimé par:

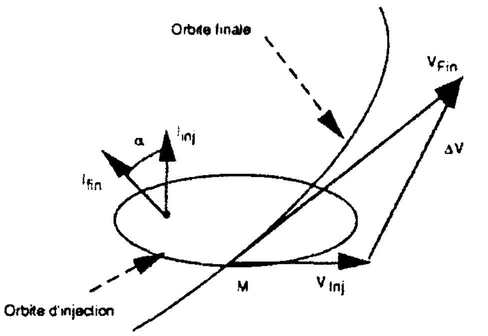
$$V_o = \sum_i M_{ergi} \quad \text{ou} \quad V_o = \underset{i}{Max} (M_{ergi})$$

Intéressons-nous maintenant le calcul de l'impulsion de vitesse dans le cas de manoeuvres mono ou bi-impulsionnelles.

2.3.2 Calcul des manoeuvres

* Manoeuvre mono-impulsionnelle

Afin de déterminer où s'effectue la manoeuvre sur l'orbite d'injection, un paramètre supplémentaire à optimiser est pris en compte: le temps séparant l'injection de la manoeuvre.



- Figure 1 -

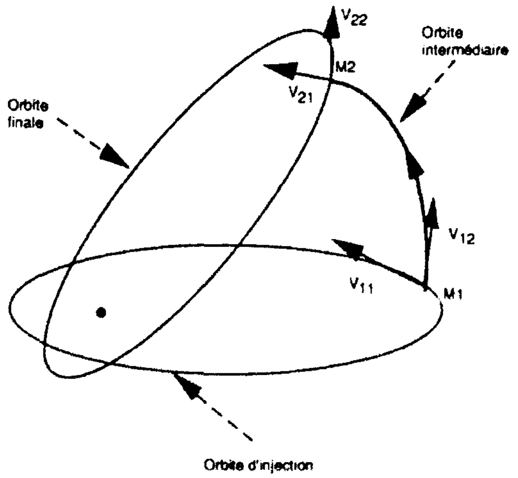
I_{inj} et I_{fin} sont respectivement les vecteurs inclinaisons sur l'orbite d'injection et finale. V_{inj} et V_{fin} sont respectivement les vitesses sur l'orbite d'injection et sur l'orbite finale au point de manoeuvre M (voir figure 1).

Pour assurer que le point M appartient précisément à l'orbite finale désirée, deux contraintes supplémentaires doivent être ajoutées concernant l'altitude et la latitude de M. Si elles ne sont pas satisfaites, l'orbite finale est modifiée de manière à pouvoir calculer le coût (voir §2.4.1).

Après la modification éventuelle de l'orbite finale, la vitesse V_{fin} et donc ΔV sont facilement calculables.

* Manoeuvre bi-impulsionnelle

Contrairement à la manoeuvre mono-impulsionnelle, une manoeuvre bi-impulsionnelle est toujours possible, mais requiert un plus grand nombre de paramètres à optimiser.



- Figure 2 -

Le satellite est injecté en M_1 depuis l'orbite d'injection sur une orbite intermédiaire sécante à l'orbite finale. Au point d'intersection M_2, une seconde impulsion est donnée pour placer le satellite sur son orbite finale (voir figure 2).

Afin de définir complètement ce transfert, trois paramètres sont optimisés:

- *un paramètre donnant la position de M_1 sur l'orbite* d'injection. On choisit l'ascension orbitale vraie, qui est l'angle brisé:

$$\psi_1 = \Omega_1 + \omega_1 + v_1$$

où Ω_1 est la longitude du noeud ascendant, ω_1 est l'argument du périgée, et v_1 l'anomalie vraie du satellite sur l'orbite d'injection.

- un paramètre donnant la position de M_2 sur l'orbite finale, ψ_2

- un paramètre déterminant la forme de l'orbite intermédiaire, joignant M_1 et M_2. Par deux points non alignés avec le centre de la Terre, passe seulement une orbite Képlérienne de paramètre fixé p_T. p_T sera choisi comme troisième paramètre.

Deux autres paramètres pourront aussi être pris en compte (*s'ils ne sont pas contraints*): *l'argument du périgée et du* noeud ascendant de l'orbite finale.

Connaissant la position de M_1 et M_2 et le paramètre p_T, il est possible de déterminer les autres éléments de l'orbite intermédiaire, nécessaires pour le calcul des vitesses V_{12} et V_{21}, respectivement aux points M_1 et M_2. En particulier, il existe deux solutions i_T et $-i_T$ pour le vecteur inclinaison de l'orbite intermédiaire, correspondant aux deux sens de parcours, et donc changeant V_{12} et V_{21} en leurs opposés. La solution adoptée sera celle conduisant à une impulsion de vitesse minimale.

2.4 Contraintes

Les contraintes à respecter concernent aussi bien la trajectoire de montée que les diverses manoeuvres. Elles sont donc de différents types.

2.4.1 Contraintes finales

* Contraintes sur le lanceur

Les contraintes finales sur le lanceur permettent d'assurer que l'orbite d'injection est atteinte au temps final (déterminé par l'épuisement des ergols). Une première contrainte d'inégalité porte sur l'altitude du périgée de l'orbite d'injection afin qu'elle soit viable pendant plusieurs révolutions ($h_p \geq 200$ km).

* Contraintes sur les satellites

Un jeu de contraintes d'égalité impose le respect des éléments orbitaux finaux (a, e, i et éventuellement Ω, ω et v) des différents satellites.

Des contraintes d'inégalité supplémentaires sont activées selon le type de manoeuvre.

Dans le cas d'un transfert mono-impulsionnel, deux contraintes sont nécessaires pour assurer que les orbites d'injection et finale sont sécantes au point de manoeuvre M, contraintes portant sur l'altitude et la latitude de ce point.

Si h_M est l'altitude de M, h_{pF} et h_{aF} respectivement l'altitude du périgée et de l'apogée de l'orbite finale, on doit avoir:

$$h_{pF} \leq h_M \leq h_{aF}$$

Si cette contrainte n'est pas respectée, l'altitude du périgée est modifiée comme suit:

$$h_{pF} = 2(a_F - R_T) - h_M \quad \text{(où } R_T \text{ est le rayon terrestre)}$$

Si ϕ_M est la latitude de M, i_F l'inclinaison finale, on doit avoir:

$$|\phi_M| \leq i_F \text{ (ou } |\phi_M| \leq \pi - i_F \text{ si } i_F \geq \pi/2)$$

Si ça n'est pas le cas i_F est modifiée en $i_F' = |\phi_M|$
(ou $i_F = \pi - |\phi_M|$ si $i_F \geq \pi/2$)

Dans le cas d'un transfert bi-impulsionnel, seule une contrainte d'inégalité est ajoutée pour assurer la viabilité de l'orbite intermédiaire (altitude de périgée supérieure à 200 km).

2.4.2 Contraintes courantes

Une seule contrainte courante sera prise en compte. Elle porte sur la trajectoire ascensionnelle du lanceur, et assure que le flux thermique maximal reçu par la charge utile après le largage de la coiffe n'excède pas 1135 W/m^2. Le flux thermique est modélisé de façon simplifiée par:

$$\phi = 1/2\, \rho\, V_a^3$$

où ρ est la densité atmosphérique et V_a la vitesse aérodynamique du lanceur.

2.4.3 Contraintes intermédiaires

Les contraintes intermédiaires prévues dans le code, telle que la contrainte de retombée d'un étage ([2]), ne seront pas prises en compte dans les cas étudiés afin de simplifier le problème.

3 RESULTATS

L'outil présenté précédemment permet de trouver la "meilleure" trajectoire de montée (au sens d'un coût) ainsi que les "meilleures" manoeuvres, grâce à une optimisation couplée. De plus, il est possible de considérer la mise à poste simultanée de plusieurs satellites sur des orbites différentes.

La capacité du code à traiter des problèmes variés sera démontrée sur les deux exemples suivants: le premier concerne deux satellites à injecter sur des orbites d'inclinaisons différentes et le second la mise à poste d'une constellation de trois satellites sur des orbites de même inclinaison mais dont les noeuds ascendants sont séparés de 120°.

3.1 Lancement double

Considérons deux satellites dont les caractéristiques sont les suivantes:
- satellite 1 : masse: 2500 kg (moteur d'apogée compris)
  orbite visée : 700/700/98°
- satellite 2 : masse: 1000 kg (moteur d'apogée compris)
  orbite visée : 700/700/80°

Le lanceur considéré est Ariane 44LP équipé de sa coiffe longue (815 kg), du support externe Speltra (420 kg) et de l'adaptateur Ariane (50 kg).

Plusieurs solutions peuvent être envisagées pour effectuer ce lancement double.

La première consiste à injecter directement l'un des satellites sur son orbite finale et d'effectuer une manoeuvre pour le second. Dans ce cas, la masse nécessaire pour cette manoeuvre peut dépasser les capacités d'emport des moteurs d'apogée existant actuellement.

Une autre solution peut donc être d'injecter les deux satellites sur une orbite de transfert et d'effectuer une manoeuvre pour chaque satellite. Cette orbite d'injection sera optimisée au sens d'un coût (par exemple, la minimisation des masses d'ergols dépensés).

3.1.1 Orbite d'injection fixée

Puisque les orbites visées de chaque satellite O_1 et O_2 ont même altitude (700 km), l'impulsion de vitesse nécessaire au transfert d'une orbite à l'autre requiert uniquement un rattrapage d'inclinaison de 18°, sous réserve que la manoeuvre ait lieu à l'un des noeuds. La valeur minimale de l'impulsion est dans ce cas 2350 m/s. On suppose que les deux satellites sont équipés du même type de moteur d'apogée dont les caractéristiques sont basées sur celles de la gamme MAGE (Isp = 310 s, rapport de la masse sèche sur la masse d'ergols de 10 %).

La table 1 donne les masses nécessaires à mettre à poste sur l'orbite d'injection pour les deux types de manoeuvres (O_1 vers O_2 et O_2 vers O_1) comparées avec les performances d'Ariane 44LP.

	orbite d'injection	
Masses (kg)	O_1 (98°)	O_2 (80°)
Speltra + adaptateur	470	470
Satellite positionné	2500	1000
Satellite à positionner *	1000	2500
Masse d'ergols nécessaire	1165	2910
Masse à l'injection	5135	6880
Performance Ariane 44LP	5520	6065

* comprenant la masse sèche du moteur d'apogée

- Table 1 -

Etant donné que la masse nécessaire sur l'orbite d'injection O_2 est supérieure à la performance d'Ariane 44LP sur cette orbite, la seconde stratégie (injection sur O_2 et transfert du satellite 1 vers O_1) n'est pas viable.

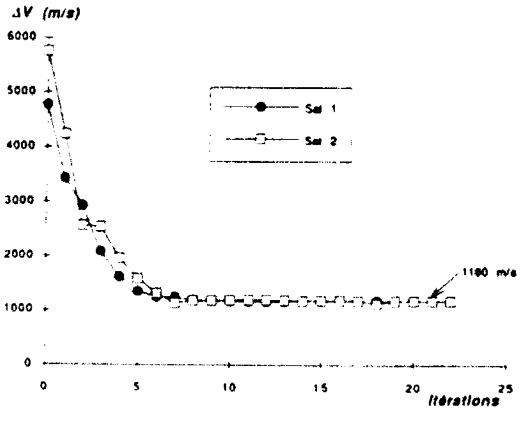
En revanche, en ce qui concerne la première stratégie, la performance d'Ariane 44LP est suffisante, mais le transfert du satellite 2 vers son orbite finale requiert un moteur d'apogée pouvant contenir 1165 kg d'ergols, moteur n'existant pas jusqu'à maintenant.

3.1.2 Optimisation de l'orbite d'injection

Le seul moyen de réaliser ce lancement double est donc d'optimiser l'orbite d'injection de telle façon que les deux satellites soient équipés de moteurs d'apogée réalistes. Deux calculs d'optimisation seront effectués avec des coûts différents.

Le premier indice de coût à minimiser est le maximum des impulsions de vitesse. Ce cas de calcul évident ne fait que démontrer les capacités du code, et par conséquent, on ne tiendra pas compte des résultats concernant l'existence des moteurs d'apogée requis. Avec cette fonction coût, l'algorithme tend à égaler les deux impulsions, en cherchant à minimiser la plus grande. Il est clair que l'orbite d'injection optimale devra être inclinée à 89°. Effectivement, l'orbite d'injection optimale O_3 trouvée par l'algorithme est la suivante: 670/720/89°. On peut noter que les manoeuvres ont lieu aux noeuds ascendants des orbites d'injection et finales, ce qui minimise le coût puisque seul un rattrapage strictement en inclinaison est à effectuer. De plus, l'orbite d'injection s'est déformée par rapport à l'initialisation (700/700/90°), diminuant son énergie En effet, puisque les manoeuvres sont mono-impulsionnelles, elles ont toutes lieu à 700 km d'altitude. L'incrément de vitesse est minimal si le module de la vitesse sur l'orbite d'injection a une valeur légèrement inférieure à celle sur l'orbite finale.

Les figures 3 et 4 présentent respectivement les évolutions au cours des itérations des impulsions de vitesses et des masses d'ergols nécessaires aux manoeuvres. On s'aperçoit qu'à partir de l'itération 50, l'optimum est presque atteint, bien que l'initialisation en soit loin.

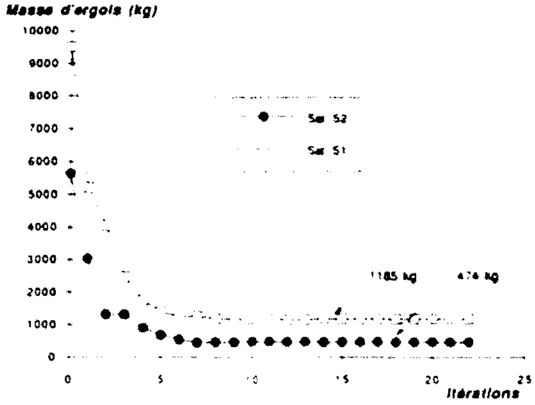


- Figure 3 -

La table 2 donne le détail de la masse nécessaire sur l'orbite d'injection O_3.

Speltra + adaptateur	Sat 1	Erg. 1	Sat 2	Erg. 2	Total
470	2500	1185	1000	474	5629

- Table 2 -



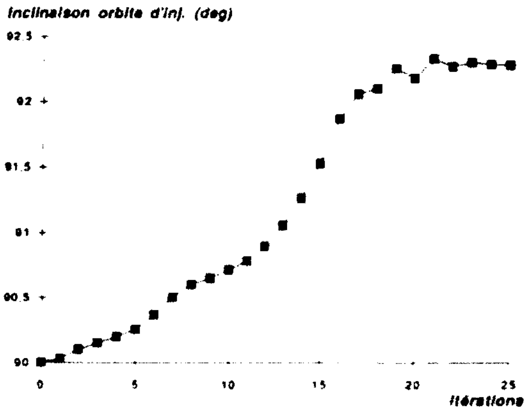
- Figure 4 -

Ce lancement est possible avec Ariane 44LP puisque sa performance sur cette orbite d'injection est de 5680 kg.

La seconde fonction de coût prise en compte est le maximum des masses d'ergols nécessaires aux manoeuvres, coût qui permet d'équilibrer les masses d'ergols des deux moteurs d'apogée. L'orbite d'injection optimale O_4 obtenue par le code est la suivante : 695/700/92.3°.

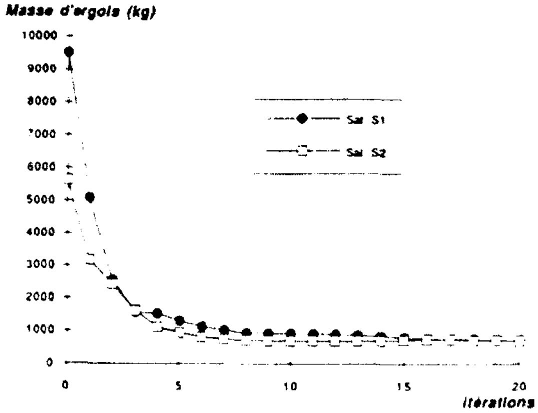
Cette valeur de l'inclinaison plus forte que précédemment est due au fait que, puisque les deux satellites doivent dépenser autant d'ergols, l'orbite d'injection est naturellement plus proche de celle du satellite le plus lourd (ici l'orbite inclinée à 98°).

La figure 5 donne l'évolution de l'inclinaison de l'orbite d'injection (initialisée à 90°) au cours ces itérations.

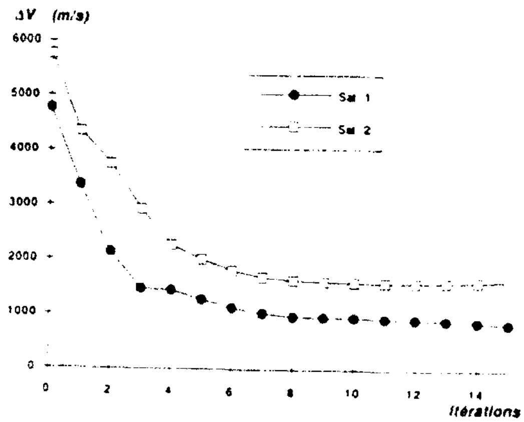


- Figure 5 -

Les figures 6 et 7 montrent respectivement les évolutions des masses d'ergols et des impulsions de vitesse.



- Figure 6 -



- Figure 7 -

A l'initialisation, les masses d'ergols nécessaires étaient de 9500 kg pour le satellite 1 et de 5650 kg pour le satellite 2. Grâce à l'optimisation de l'orbite d'injection, ces valeurs ont été considérablement réduites pour arriver à des masses réalistes (700 kg).

La table 3 fournit le détail des différentes masses sur l'orbite d'injection O_4.

Speltra + adaptateur	Sat 1	Erg. 1	Sat 2	Erg. 2	Total
470	2500	700	1000	700	5370

- Table 3 -

La capacité d'emport d'Ariane 44LP pour cette mission est de 5512 kg.

Dans le cas du lancement double, l'optimisation couplée de la trajectoire de montée et des manoeuvres a conduit à une orbite d'injection qui permet de mettre à poste les deux satellites avec des masses d'ergols réalistes.

3.2 Lancement d'une constellation

Considérons une constellation de radiolocalisation [3] composée de trois satellites, de 500 kg chacun, dont les orbites finales ont pour éléments orbitaux :

$a = 42164$ km
$e = 0{,}191$ ($h_p = 27733$ km et $h_a = 43840$ km)
$i = 18{,}345°$
$\omega = 254{,}887°$

plus une différence de noeuds ascendants de 120° entre les orbites.

Le lanceur utilisé pour ce lancement triple est Ariane V, équipé de la coiffe courte (1870 kg) et du support externe Speltra (900 kg). Les transferts sont de type bi-impulsionnel et la fonction coût à minimiser est la somme des impulsions de vitesses des trois manoeuvres. Comme précédemment, on considèrera tout d'abord une les transferts à partir d'une orbite d'injection fixée qui sera libérée par la suite afin d'observer les améliorations apportées par le code. Enfin, dans les deux cas, une contrainte supplémentaire pourra être activée afin d'assurer un bon équilibre massique entre les satellites. La satisfaction de cette contrainte moteur permet de limiter à 5% l'écart entre les masses d'ergols nécessaires à chaque satellite et leur valeur moyenne.

3.2.1 Orbite d'injection fixée

L'orbite d'injection considérée est une GTO classique inclinée à 7° d'altitude périgée 280 km et d'apogée à 35786 km d'altitude.

Choisir une orbite d'injection déjà inclinée à 18,345° peut sembler au premier abord plus intéressant en terme de coût, puisqu'il ne resterait plus qu'un rattrapage de noeud ascendant à effectuer. Néanmoins, les changements de noeud ascendant entraîneraient obligatoirement des modifications d' inclinaison (sauf pour le satellite dont l'orbite finale a le même noeud ascendant que l'orbite d'injection), ce qui conduirait à un coût très important.

Cette orbite GTO servira d'initialisation pour les optimisations suivantes.

Deux calculs ont été menés pour la mise à poste à partir de cette orbite d'injection, la contrainte moteur n'étant activée que dans un deuxième temps. Bien entendu, le coût est plus important dans ce cas (707 m/s contre 6631 m/s).

La table 4 donne les résultats de l'optimisation dans le cas où aucune contrainte n'est imposée sur les masses d'ergols. Il est à noter que les valeurs des noeuds ascendants ne sont que relatives. En effet, le temps initial correspond arbitrairement au moment où γ_{50} passe à Greenwich. Pour modifier le noeud ascendant de l'orbite d'injection, il suffit donc d'ajuster l'heure de tir.

Les masses d'ergols pour les trois satellites sont très différentes. Pour éviter ces écarts, on active la contrainte moteurs.

	Ω (°)	ΔV (m/s)	Masse d'ergols (kg)
Orbite d'injection	180	-	-
Orbite finale 1	9	1833	414
Orbite finale 2	129	2363	588
Orbite finale 3	249	2435	614

- Table 4 -

La table 5 résume les modifications induites par l'activation de cette contrainte.

	Ω (°)	ΔV (m/s)	Masse d'ergols (kg)
Orbite d'injection	180	-	-
Orbite finale 1	9	2278	558
Orbite finale 2	129	2355	585
Orbite finale 3	249	2438	614

- Table 5 -

Du fait que les masses d'ergols des deux derniers satellites étaient du même ordre de grandeur, l'algorithme n'a fait que dégrader le coût de la manoeuvre du premier satellite.

3.2.2 *Optimisation de l'orbite d'injection*

* Contrainte moteurs désactivée

Le point de départ de l'optimisation est résumé dans la table 4. En particulier, la somme des incréments de vitesse nécessaires aux transferts depuis une GTO 7° est de 6631 m/s. Après 300 itérations, elle est passée à 5934 m/s, et l'inclinaison de l'orbite d'injection est passée à 13,8° et l'argument du noeud ascendant à 147,3°. De plus, l'énergie de l'orbite a augmenté les altitudes de périgée et d'apogée passant respectivement de 280 km à 611 km et de 35786 km à 40586 km.

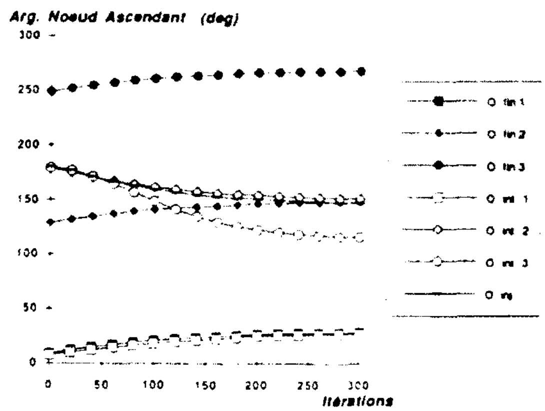
On peut noter qu'au cours des itérations, l'écart relatif des noeuds ascendants des orbites finales demeure égal à 120°, cette contrainte restant satisfaite par la suite.

La table 6 compare les orbites entre l'initialisation et la convergence.

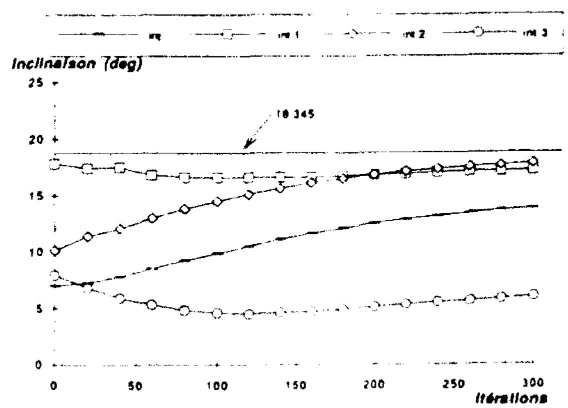
	Itération 0	Itération 300
i injection (°)	7	13,8
Ω injection (°)	180	147,3
Ω finaux (°)	9/129/249	28/148/268
ΔV (m/s)	1833/2363/2435	1737/1355/2842
Masse d'ergols (kg)	414/588/614	385/280/773
Coût (m/s)	6631	5934

- Table 6-

Les figures 8 et 9 fournissent plus de détails sur les changements intervenus au cours des itérations. Elles présentent respectivement les évolutions des noeuds ascendants et des inclinaisons des orbites d'injection, intermédiaires et finales.



- Figure 8 -

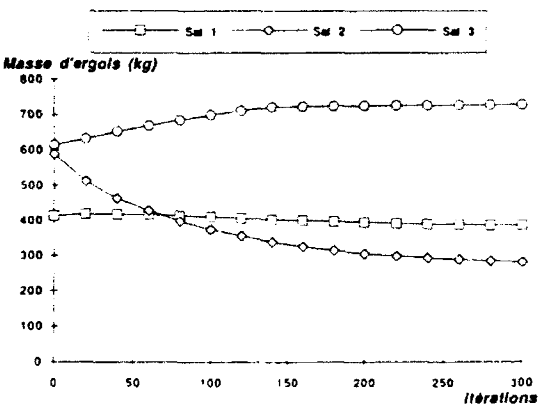


- Figure 9 -

Ces figures montrent qu'à la convergence, les orbites intermédiaire et finale du satellite 2 ont pratiquement même inclinaison et noeud ascendant, contrairement à l'initialisation. La figure 8 montre en particulier que ce noeud ascendant commun aux orbites intermédiaire et finale est aussi celui de l'orbite d'injection, ce qui explique le faible coût des manoeuvres effectuées par le satellite 2. Le transfert est en fait quasi mono-impulsionnel (1ère impulsion: 1278 m/s, 2nde impulsion: 77 m/s), la première impulsion permettant un rattrapage global en énergie et en inclinaison.

Pour le satellite 1, on constate une évolution relativement minime des orbites finale et intermédiaire entre les itérations 0 et 300, aussi bien en terme de noeud ascendant que d'inclinaison. Comme pour le satellite 2, le transfert est quasiment mono-impulsionnel (1ère impulsion: 1662 m/s, 2nde impulsion: 75 m/s).

Enfin, les écarts en inclinaison et en noeud ascendant entre les orbite d'injection, intermédiaire et finale du satellite 3 ont sensiblement augmenté au cours des itérations. Par conséquent, ce transfert est beaucoup plus coûteux que les deux autres (773 kg d'ergols requis contre 385 et 280 kg pour les satellites 1 et 2). L'évolution des masses d'ergols est illustrée sur la figure 10. On observe la diminution des masses d'ergols du satellite 2 permettant une amélioration du coût, ainsi que la dégradation de la manoeuvre du satellite 3.



- Figure 10 -

* Contrainte moteurs activée

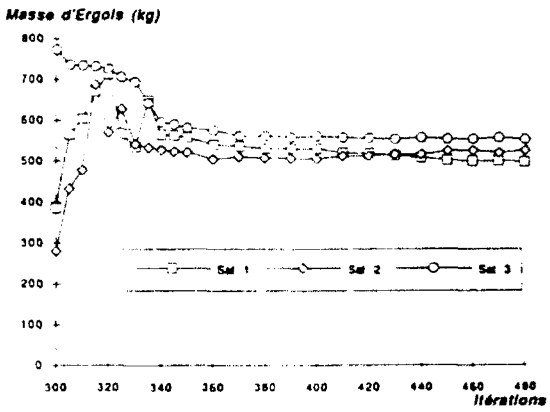
Les résultats précédents montrent une sensible diminution du coût global, mais au détriment de l'équilibre entre les masses d'ergols des différents satellites. Pour assurer la symétrie de la constellation, à partir de l'itération 300, une série de 300 itérations supplémentaires a été effectuée en activant la contrainte moteurs.

La table 7 compare les résultats des deux convergences suivant que la contrainte moteurs est active ou non. Comme on pouvait le prévoir, le respect de cette contrainte entraîne une augmentation du coût total. Cette valeur obtenue est à comparer la avec celle correspondant au cas de l'orbite d'injection fixée à une GTO 7°, contrainte moteur active (§3.2.1), où l'on constate une nette amélioration, le coût passant de 7071 m/s à 6526 m/s.

	Itération 300	Itération 600
i injection (°)	13,8	6,4
Ω injection (°)	147,3	171,2
Ω finaux (°)	28/148/268	-19/100/220
ΔV (m/s)	1737/1355/2842	2095/2178/2253
Masse ergols (kg)	385/280/773	496/524/549
Coût (m/s)	5934	6526

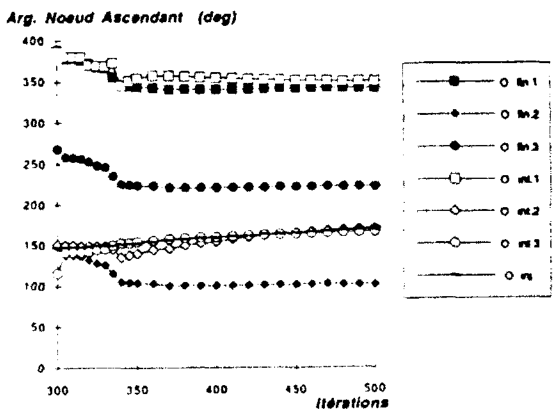
- Table 7 -

La figure 11 donne les évolutions des masses d'ergols de chaque satellite au cours des itérations.
On note que seules 50 itérations sont nécessaires à la satisfaction de la contrainte moteurs, les masses d'ergols requises gardant quasiment les mêmes valeurs par la suite.

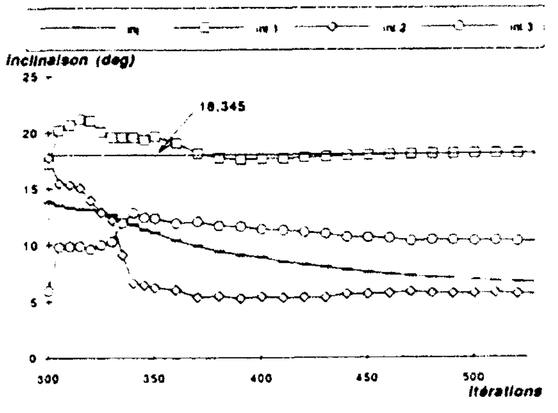


- Figure 11 -

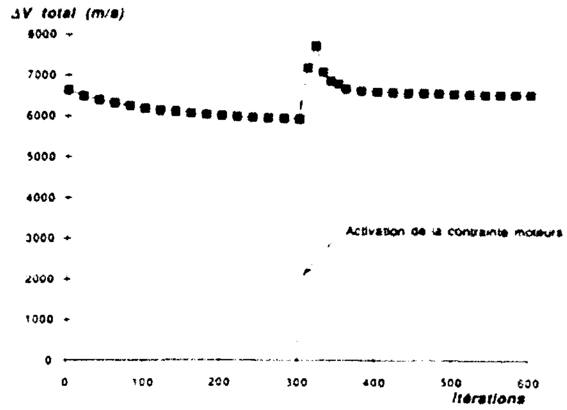
Les figures 12 et 13 décrivent respectivement les évolutions des noeuds ascendants et des inclinaisons des orbites d'injection, intermédiaires et finales.



- Figure 12 -



- Figure 13 -



- Figure 14 -

Les orbites intermédiaire et finale du satellite 1 ont même inclinaison et noeud ascendant, et ont très peu varié par rapport à l'itération 300. En revanche, l'incrément de vitesse a augmenté du fait de la plus grande différence entre les inclinaisons des orbites d'injection et intermédiaire (voir Figure 13).

En ce qui concerne le satellite 2, étant donné que son transfert était auparavant le moins coûteux, l'activation de la contrainte ne pouvait que le dégrader. Ainsi, alors qu'à l'itération 300 les orbites intermédiaire et finale avaient même inclinaison et noeud ascendant, les 300 itérations supplémentaires ont conduit à une différence de 72° sur les noeuds ascendants et 12,8° sur les inclinaisons. En conséquence, et de manière à rattrapper ces écarts, la seconde impulsion n'est plus négligeable (le transfert est cette fois réellement bi-impulsionnel). Le coût du transfert est donc mieux réparti entre les deux impulsions.

En revanche, la symétrie entre les trois satellites imposée par la contrainte moteurs conduit à une baisse du coût de transfert du satellite 3. Comme le montre la figure 12, dès les 50 premières itérations, les orbites d'injection et intermédiaire ont même noeud ascendant, qui par la suite se rapproche de plus plus de celui de l'orbite finale. Quant aux inclinaisons, celle de l'orbite intermédiaire se rapproche, au cours des itérations, à la fois de l'orbite d'injection et de celle de l'orbite intermédiaire.

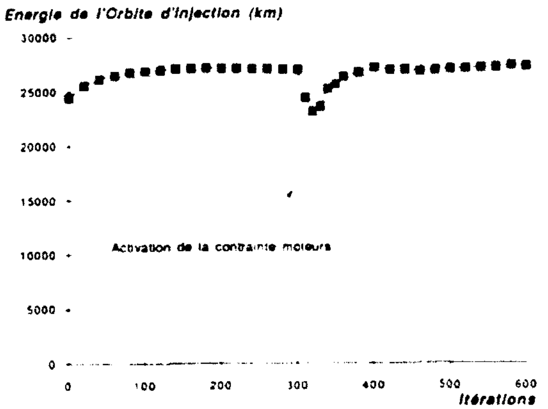
Bien évidemment, le respect de la contrainte moteurs impose que les masses d'ergols des trois satellites soient sensiblement identiques, comme l'illustrent les valeurs portées dans la table 7.

Les figures suivantes présentent des évolutions globales au cours des 600 itérations effectuées, le raccord à 300 itérations marquant l'activation de la contrainte moteurs.

La figure 14 décrit l'évolution de la fonction coût (c'est-à-dire de la somme des impulsions de vitesse).

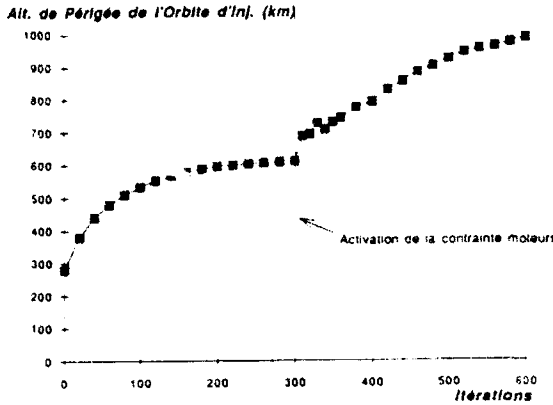
Le coût décroît régulièrement jusqu'à l'itération 300, toutes les contraintes étant déjà satisfaites. Puis, à l'activation de la contrainte moteurs, il augmente rapidement puis décroît à nouveau régulièrement une fois que la contrainte est satisfaite. La valeur trouvée après 600 itérations est bien entendu supérieure à celle de l'itération 300.

Les figures 15 et 16 donnent respectivement l'évolution de l'énergie et de l'altitude de périgée de l'orbite d'injection.



- Figure 15 -

Dans la première partie (de 0 à 300 itérations), les valeurs de l'altitude de périgée et de l'énergie de l'orbite d'injection croissent afin de se rapprocher de celles des orbites finales, permettant ainsi au coût de diminuer. Lorsqu'on active la contrainte moteurs, l'énergie décroît fortement, en parallèle avec l'augmentation du coût. Cette baisse d'énergie permet de dégrader les manoeuvres les moins coûteuses (satellites 1 et 2). Cependant, elle n'empêche pas pour autant l'amélioration du transfert du satellite 3, due essentiellement à la diminution des rattrapages en inclinaison et en noeud ascendant à effectuer (voir figures 12 et 13). Une fois la contrainte satisfaite, la valeur de l'énergie revient à son niveau précédent, et l'optimisation se poursuit avec une diminution du coût global. Notons que l'altitude du périgée de l'orbite d'injection n'a cessé d'augmenter au cours des itérations, favorisant ainsi une baisse du coût, au détriment de la masse totale de charge utile placée sur cette orbite.



- Figure 16 -

4 CONCLUSION

Ces deux exemples ont montré l'intérêt d'optimiser de façon couplée la montée et les manoeuvres dans le cas de lancements multiples.

Dans le cas du lancement double, l'optimisation a permis d'effectuer la mise à poste de deux satellites d'observation au moyen de moteurs d'apogée réalistes.

Pour le lancement de la constellation, on a pu gagner, grâce à l'optimisation, entre 7 et 10% par rapport à la mise à poste depuis une GTO 7° classique.

Grâce à la diversité des contraintes -facilement activables- et aux nombreuses fonctions coûts, un grand nombre de cas peuvent être traités par le programme. L'intérêt de cette méthode d'optimisation repose sur le fait qu'elle peut gérer un grand nombre de paramètres (particulièrement dans le cas des manoeuvres bi-impulsionnelles), évitant ainsi un balayage fastidieux de toutes les combinaisons possibles.

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[2] Ph. Landiech, C. Aumasson: "Montée optimale du lanceur Ariane V-L5 avec contrainte de retombée du premier étage." R.T. n° 21/6115 SY - ONERA 1986

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Discussion

Question: My question concerns the equivalence constraint you use in order to limit to 5% the gap between each satellite propellant mass and their average value. Is it possible to minimize the maximum value of the propellant masses instead of activating this constraint?

Reply: The cost index (i.e., minimizing the maximum of the propellant masses) may be considered as it is presented in the paper, and could have been used in this case instead of trying to minimize the criterion (sum of the delta v) while activating the equivalence constraint. Never the less, the 5% margin allows a respective balance between the three satellite masses without making them <u>exactly</u> equal (which might be too strong a constraint in this complex system where the launcher ascent phase and three bi-pulse maneuvers are simultaneously optimized).

QuickStar
System Design, Capabilities and Tactical Applications of a Small, Smart Space System

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SUMMARY

The QuickStar System is not a concept—it is a flight-proven design. QuickStar is a small highly capable, low-cost, light-weight spacecraft using modern design techniques that can carry tactical assets into space at a tenth of the cost of current much larger and more costly space systems. System design, relevant technologies, payload capabilities (mass, power, data rate, pointing, volume), and tactical mission applications of this light-weight satellite are described. As part of the overall small satellite system architecture, a portable, low-cost multi-purpose ground station to support production, test, launch, and orbit operations of the space segment is also available. The versatility and transportability built into the ground station allows placement at any government installation or field site.

The prototype QuickStar space segment was developed to ride as a secondary payload on a McDonnell Douglas Delta II series expendable launch vehicle (ELV). With the extra performance provided by the Delta II or other ELV of similar configuration, as many as four satellites could be orbited at one time. In addition, the paper describes a QuickStar satellite configuration, incorporating the same subsystems and capabilities as the prototype QuickStar, designed for stand-alone ELV (Scout or Pegasus) launchings.

The program schedule for the design, fabrication, and test of a QuickStar satellite system reflects the fast-paced environment of a low-cost program. Minimal paper and much concurrent engineering goes into a schedule that provides a flight-ready spacecraft and supporting ground station in only 15 months.

1.0 INTRODUCTION

The QuickStar program represents a unique opportunity to perform critical missions while using the excess capabilities of upcoming McDonnell Douglas Delta II launches. QuickStar, "The Complete System", provides a highly capable, low-cost spacecraft incorporating a flight-proven design for fast, reliable access to space at launch costs much below the expected norm.

The QuickStar design presented in this paper incorporates existing but advanced design concepts from a number of contracted and internally funded programs at Ball Space Systems Division (Ball). In addition to the QuickStar space vehicle, this paper also includes the development of a ground station to serve as the primary command and control center.

2.0 BACKGROUND

Small, smart systems can revolutionize the way government and commercial interests implement research, strategic, and even tactical space systems. To date, the implementation of such systems has been inhibited by the widely held perception that small systems are not likely to be useful, reliable, or cost-effective. This negative perception about small systems can only be corrected by a systematic approach which addresses specific concerns and promotes mission utility. To achieve this end, Ball Aerospace in 1988 initiated an internally funded effort called TECHSTARS. The objective of the TECHSTARS program was to examine the system architectures associated with small systems and to incorporate the advanced technologies required for the 1990s to produce light-weight space systems that are highly capable and yet can be produced at low cost.

QuickStar is a program that uses the approach and knowledge gained from the TECHSTARS effort.

3.0 DESIGN APPROACH

The QuickStar design is a derivative of the U. S. Government-funded prototype spacecraft, LOSAT-X. **Figure 3-1** is a picture of LOSAT-X in the clean room at Ball just prior to shipment to Cape Canaveral Air Force Station Delta Launch Complex 17. LOSAT-X was the result of a government push to develop, test, launch, and operate small spacecraft and complementary sensor technologies. Designed by Ball Aerospace, the LOSAT-X spacecraft included an integrated avionics suite built around two 80C86 processors, a 0.25-Gbit mass memory, Ball-developed reaction wheels, and a new wide field-of-view (WFOV) star camera. Design drivers dictated that this complicated spacecraft fit within a very small envelope on a McDonnell Douglas Delta II rocket as a secondary payload and still be sophisticated enough to accomplish mission science objectives.

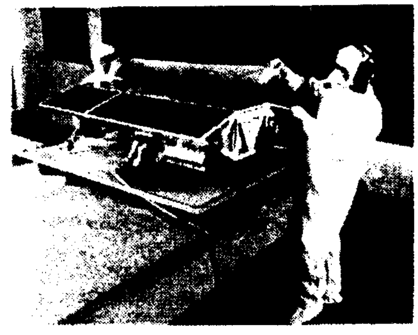

Figure 3-1 The LOSAT-X spacecraft

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*Chief Engineer, LOSAT-X/QuickStar
+Asst. Vice President, Business Development

4.0 LAUNCH SYSTEM OVERVIEW

As with the case of LOSAT-X, QuickStar is launched as a secondary payload on a Delta II. The secondary payload concept is shown in **Figure 4-1**. The QuickStar spacecraft is attached to the Delta second stage through a payload adapter system positioned between the second stage guidance section and the vehicle fairing, well outside and below the primary payload envelope. Depending on the excess launch capability of the particular Delta II launch, up to four QuickStars could be carried into space on a single launch. The merit of not requiring a dedicated launch substantially reduces launch costs and provides more launch opportunities to the user. However, if required, QuickStar is easily adaptable to other launch vehicles because of its small size and light weight.

5.0 SPACECRAFT SYSTEM

QuickStar is a free-flyer. Design life is six months with a goal of up to one year. QuickStar is a small highly capable, low-cost, light-weight satellite system using modern design techniques. The QuickStar spacecraft is a 3-axis stabilized vehicle that uses three reaction wheel assemblies, three magnetic torque rods, a star tracker, and a 3-axis gyro package to maintain attitude control and determination. **Figure 5-1** is a system concept summary and illustration of the QuickStar spacecraft. The QuickStar satellite performance exceeds typical small satellite capabilities (**Figure 5-2**).

A functional block diagram of the QuickStar flight system shown in **Figure 5-3** illustrates the extent that the spacecraft is under software control. Through the use of an integrated central processor, as compared to most satellite systems that are a combination of dedicated hardware control units and software processors, QuickStar is able to improve reliability by replacing hardware with software at the same time reducing volume, power, and weight requirements. In addition, with the integrated central processor, extensive testing of all spacecraft systems and control modes is possible on the ground providing the confidence that it will function the same way on orbit.

Spacecraft equipment is mounted inside of the spacecraft or attached to the exterior structure within the envelope provided (Figure 4-1). Equipment attached to the spacecraft exterior includes three avionics modules, three solar array panels, reaction wheels, torque rods, two patch antenna sets, trickle charge and test connectors, and a separation fitting/connector. Internal equipment includes a communication transponder, gyro, battery, and payload. **Figures 5-4** and **5-5** illustrate both internal and external component general arrangements.

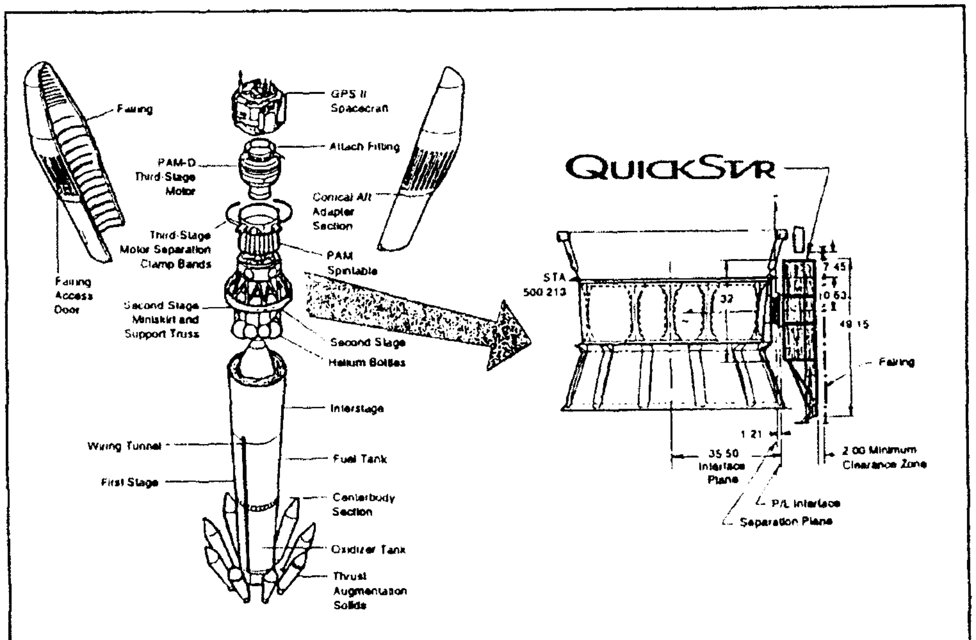
Structure/Mechanisms Subsystem

The QuickStar spacecraft structure is approximately 49 inches long by 36 inches wide and 12 inches deep. The upper deck doubles as solar panel substrates and is covered with solar cells providing approximately 12 square feet of solar array area on three panels. The lower deck assembly provides mounting surfaces for all electronics boxes and spacecraft components. The upper and lower decks are connected by four yoke assemblies forming the spacecraft bus enclosure.

A separation attach fitting is externally mounted on the lower deck. The attach fitting mates with the launch vehicle separation mechanism. Separation mechanisms and ordnance consisting of four arrayed explosive bolt/spring assemblies are provided on the launch vehicle side of the separation interface.

Thermal Subsystem

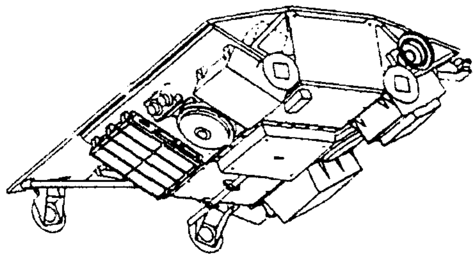
The thermal subsystem provides the capability to maintain all spacecraft components to within prescribed temperature limits. Thermal control is accomplished by passive means using high emissivity/low absorptivity finishes, and thermal isolating hardware. The spacecraft design does not require the use of multi-layer insulation blankets, louvers and/or active heaters. Telemetry provides temperature measurements from thermistors located near or on various key components and payload equipment.



McDonnell Douglas Delta launch Vehicle with GPS Spacecraft/QuickStar

Figure 4-1 Integration of QuickStar to the Delta II Launch Vehicle

GENERAL ARRANGEMENT



KEY FEATURES

- Delta II compatible
- Spacecraft Mass = 150.0 lbm (68.2 kg)
- Payload
  - Mass = 70.0 lbm (31.8 kg)
  - Volume = 1.7 ft^3(0.05 m^3)
  - Extended Vol. = 0.8 ft^3(0.02 m^3)
- 3-axis stabilization
- Power
  - 130 W (average)
  - 214 W (peak)
- SGLS compatible communication links
  - Uplink 1 Kbps
  - Downlink
    - Payload: 1.0 Mbps
    - Spacecraft: 1.6 or 32 Kbps
  - Encrypted links, if required
- Integrated avionics
- Passive thermal control

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Figure 5-1 QuickStar spacecraft system concept

CHARACTERISTIC	USUAL SMALLSAT LIMITATIONS	QUICKSTAR CAPABILITIES	COMMENTS
Prime structure	Custom structure	Honeycomb decking/ trusswork box	Inexpensive and simple to build.
Pointing/stabilization	Inertial spinner or gravity gradient	All-pointer (3-axis)	Controlled by three 1 N-m-s reaction wheels.
Pointing control	1-3 deg	0.1 deg	
Attitude knowledge	0.25 deg	<0.1 deg	
Slew rates	No capability	5 deg/sec (3-axis)	10 deg/sec maximum.
Peak Power	<100 watts	214 watts	GaAs solar arrays.
On-board processing	500 KIPS	2 MIPS	80386 processor
Data storage	100 Mbits	1 - 2.5 Gbits	Solid state memory.
Downlink data rate	32 Kbits/sec	1 - 5 Mbits/sec	All links encrypted, if req'd.
Propulsion	Rarely considered or available as an option.	Available.	ΔV = 200-300 ft/sec.

Figure 5-2 QuickStar key subsystem capabilities

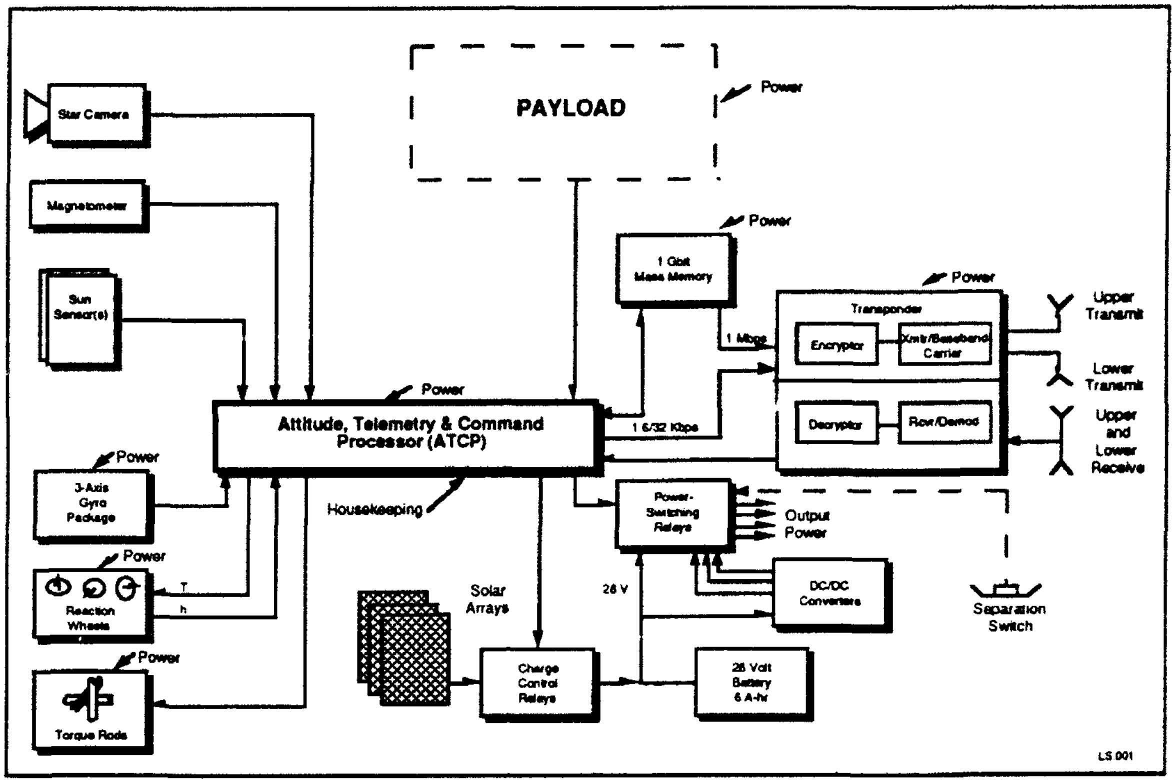


Figure 5-3 QuickStar spacecraft functional description

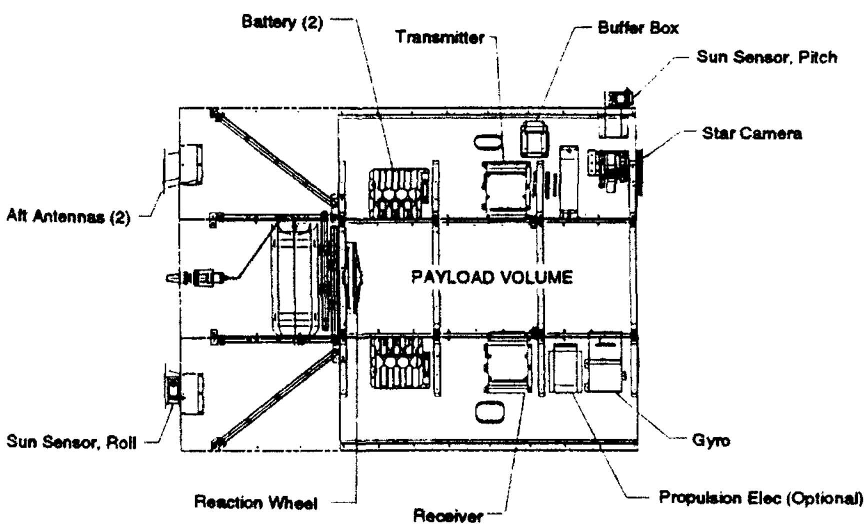


Figure 5-4 QuickStar internal component general arrangement

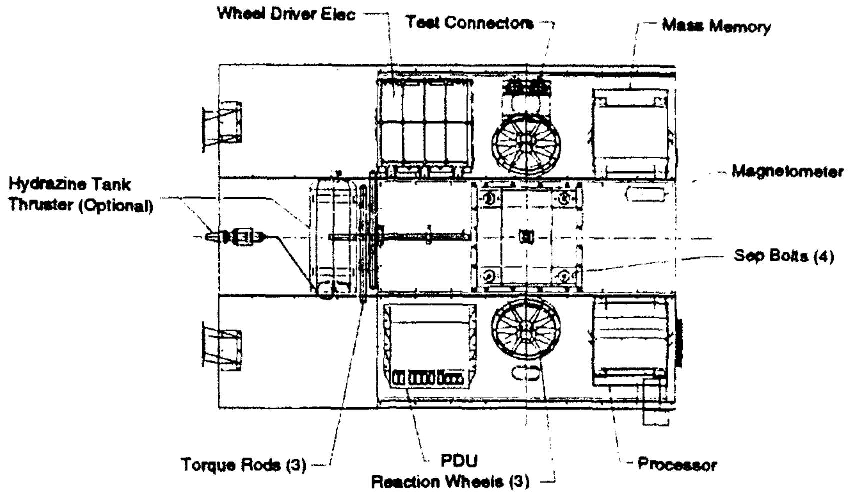


Figure 5-5 QuickStar external component general arrangement

Electrical Power Subsystem

The electrical power subsystem provides electrical bus power to energize spacecraft loads in all phases of the orbit. It provides regulated power to the payload, power switching for non-essential components, and undervoltage and overcurrent protection for the spacecraft bus. The QuickStar spacecraft uses a GaAs solar array for power collection and a battery for energy storage. The electrical power subsystem consists of three solar panels, a 6 Amp-hour NiCd battery (or an optional 12 Amp-hour battery), dc/dc converters, charge control and power switching relays. The power subsystem operates at a nominal 28 volt dc. The system is designed to provide an average of 130 watts to the spacecraft system including a 15-watt reserve.

Bus undervoltage levels ("yellow" and "red") are ground selectable. When bus voltage drops below the "yellow", non-essential components are switched off including the payload, mass memory, and spacecraft transmitter. The receiver, flight processor, gyro, PDU, reaction wheels and drivers are maintained on the essential bus. In the event of a "red" undervoltage, only the spacecraft flight processor, power distribution unit, and receiver are left powered. This configuration is sufficient to safe the spacecraft until a plan is generated by the satellite operations crews to analyze the anomaly and command disconnected equipment back on. Protection is also provided should an overcurrent condition occur. Components on the non-essential bus are switched off similarly to the bus undervoltage situation.

Avionics

The concept of an integrated avionics suite is borrowed from current developments for fighter aircraft where all the monitoring, control, housekeeping, and processing functions are integrated together, not only for the synergistic effect, but also to reduce cost and improve reliability and maintainability.

In keeping with the concept of integrated avionics, QuickStar has replaced the separate boxes for each major on-board function (i.e., command and data handling, attitude determination and control, and telemetry, track, and control) each typically with its own power supply, packaging, connectors, and cable harness with a 80386 central processor, memory, and architecture technologies.

The attitude, telemetry, and command processor (ACTP) I/O provides for 32 analog inputs and 4 outputs, 32 parallel inputs and 32 outputs, 4 serial inputs and 4 outputs, and 32 relay driver outputs. The special functions interface contains a serial I/O DMA controller, torque rod drivers, sun sensor preamplifiers, momentum wheel tachometers, transponder interface, and real-time clock circuits. The memory devices, built with CMOS technology, are packaged using state-of-the-art memory module manufacturing techniques, mounted to four printed circuit wiring boards using surface mount technologies, and architected into the avionics suite with error detection and correction circuitry. The result is a processor box that can recover from single event upsets (SEUs), is latchup free, and can tolerate high radiation levels.

Command, Telemetry, and Ranging (C,T&R) Subsystem

The CT&R subsystem provides for the communication between the spacecraft and the ground-based stations in addition to providing an interface with all spacecraft subsystems. Omnidirectional uplink capability is provided by the fore and aft L-band patch antenna sets, a communications transponder, and the spacecraft attitude, telemetry and command processor (ATCP). The command uplink rate is 1 to 2 Kbps. Downlink capability is provided by the ATCP, data memory assembly (DMA), transponder, and fore and aft S-band patch antennas. Downlink rates include 1 Mbps for the payload data and real time 1.6/32 Kbps for spacecraft health and status.

A hardware command buffer in the ATCP holds up to 512 commands which are either executed immediately upon arrival from the receiver or stored for delayed execution in command stored memory (CSM). Pseudo random noise (PRN) ranging data is also sent to the receiver via the antenna and is filtered and routed to the transmitter for turnaround transmission to the ground station (used for range determination). The receiver portion of the transponder is always powered-on from the spacecraft essential bus.

Attitude, Determination and Control Subsystem (ADACS)

The ADACS provides attitude determination and control for the QuickStar spacecraft. The ADACS consists of a star camera, a 3-axis gyro package, two sun sensors, a 3-axis magnetometer, three reaction wheel assemblies and drive electronics, and three torque rods. Attitude determination is accomplished using inputs from a small, wide-field-of-view (WFOV) solid-state star camera (via the ATCP and DMA) when in the inertial pointing mode and the 3-axis gyro package during tracking operations. Attitude control is achieved using the reaction wheels while magnetic torquing is used to dump stored momentum in the wheels.

The WFOV star camera uses recently emerging technologies in WFOV lenses and focal plane flattening. The result is a greatly simplified and lower cost star camera. Using the more capable star camera in lieu of Earth and sun sensors results in a simplified attitude determination and control subsystem. With the relatively simple charged coupled device (CCD) star camera along with a complex software program, the five brightest stars are sensed in the field-of-view and matched to an on-board star catalog to compute an attitude solution. Using the new WFOV camera, the ADACS can provide attitude knowledge to better than 0.1 degrees in all axes. In addition, its WFOV capability allows the on-board star catalog to be under 500 stars, thus minimizing on-board storage and power requirements and reducing processor loading.

During normal operations, QuickStar is maintained in a Sun Point Mode in which the solar panels are positioned normal to the sun-line for optimum power output. In this mode the star camera is used for primary attitude determination. During certain predetermined times, however, the spacecraft can be commanded to one of several pointing mode types such as "Track" or "Inertial". In "Track", the spacecraft slews so that the +X axis tracks a predetermined Earth fixed or orbital position. During this time the star camera data is unavailable so spacecraft attitude is determined by using the gyro as a reference. Attitude uncertainties continue to accumulate due to gyro drift until the end of the slew and a star camera update becomes available. The Track mode operates with any geodetic location whether on the face of the Earth or at orbital altitude. In "Inertial", the spacecraft is pointed along a commanded fixed inertial vector. During this mode, the gyro maintains zero body-axis rates. After each "Track" or "Inertial" maneuver, QuickStar is returned to "Sun Point".

Back-up attitude determination is available using a 3-axis magnetometer and the two sun sensors. Using sensed Earth field line and sun directions, spacecraft body-axis attitude solutions with accuracies on the order of 1 to 2 degrees are available.

Propulsion Subsystem

An optional element of the QuickStar system, based on mission/payload requirements, is a lightweight propulsion subsystem designed to perform delta-velocity maneuvers to effect either orbit altitude or orbit phasing. The design of the propulsion system includes a 5.1-inch diameter by 12-inch long propellant tank operating at 500 psig, a 5 lb_f dual seat thruster, an electronics box containing valve drive amplifier circuitry, and assorted fill values, pyro values, and plumbing. The propellant tank contains 6.6 lbm of hydrazine, enough propellant to supply delta-velocity capabilities on the order of 200 to 300 feet-per-second. Total propulsion subsystem weight is approximately 15 pounds including 0.4 pounds of pressurant. The thruster is positioned under the center solar array panel, canted so the thrust vector passes through the spacecraft system center-of-mass.

The QuickStar propulsion capability is operated in a "pulse" mode such that in between each thruster firing, enough time is provided for the control system to remove any off-axis disturbances that may have been induced by thrust vector misalignments.

6.0 GROUND SYSTEMS

As part of the overall small satellite system architecture, a low-cost multi-purpose ground station to support production, test, launch, and orbit operations of QuickStar has been developed. **Figure 6-1** shows the ground station providing independent data acquisition and mission control for the QuickStar missions. The versatility and transportability built into the ground station allows placement at any government installation, university, or contractor facility. Command and control of the satellite and its payload can be as close as the desk in your office.

Figure 6-1 QuickStar operational ground station

The QuickStar ground station and independent payload processing center provides support for mission planning, command generation/uplink, data acquisition, processing, and analysis. Ground station performance is summarized in **Figure 6-2**.

The QuickStar ground station consists of an RF rack housing a Global Positioning System (GPS) station clock, a digital ranging receiver, a telemetry demodulator, bit synchronizer, baseband assembly unit, signal generator, and 250-watt uplink power amplifier. A rack mount workstation style desk houses the baseband receiver, 1.0 Mbps demodulator/bit-synchronizer, antenna control unit, and two redundant 25 Mhz 80486 microprocessors. A roof mounted 2.9 meter antenna system includes the low noise preamplifier and RF downconverter. Both the 80486 processors are capable of real-time telemetry processing/display or command and equipment control.

PARAMETER	GROUND STATION PERFORMANCE
Telemetry Acquisition	
- Frequency	• SGLS, STDN and DSN Channels
- Modulation Types	• PM, BPSK, QPSK (NRZ-L,N,S)
- Data Rates	• 1 - 5 Mbps, 1.6 & 32 Kbps
- Link Margin	• 3.2 dB, 1 Mbps @ 5° Elevation
	• 14.6 dB, 32 Kbps @ 5° Elevation
Command Uplink	
- Frequency/Modulation	• Full SGLS Coverage (1760 to 1840 MHz), NASA STDN/DSN (2025 to 2120 MHz)
- Cmd Rate	• 1, 2 & 10 Kbps
- Ternary Cmd Uplink	• SGLS (TOR-59) or STDN compatible
- Link Margin	• 18.1 dB (250 W Uplink Transmitter)
Ranging & Orbit Determination	• SGLS (TOR-59) or STDN compatible
	• 1.0 Km or better (Osculating)
Antenna System	
- Gain	• 34 dBi (9.5 foot dish)
- G/T	• 12 dB/K
- Downconverter I/F	• 70 MHz ±100KHz
- Pointing Capability	• Az = 0 to 360°
	• El = 0 to 180°
	• Accuracy = 0.5° to Track
Operations Interface	
- STOL	• Syntax Duplicated
- RT Tlm Displays	• User Definable Tabular/Graphics Display
- Tlm Limits Checks	• Data Base Definable Raw Tlm Limits Chk
- Operator Alert Msg	• Provided for Limits, Cmd & System Errors
- Data Capture/Quality	• Frame Sync, Major & Minor Frame Count
- Printout Capability	• Provided by Operator or Procedure Call
- Tlm Simulator	• Major, Minor Frame, Subcom & Fixed Tlm
- Cmd Echo Validation	• Ternary Cmd Turnaround & S/C Tlm Echo

Figure 6-2 Ground station and control center performance

The complete workstation incorporates additional 80486 microprocessors and off-the-shelf equipment in a distributed processing environment providing mission planning and orbit determination support, as well as off-line data processing and analysis. Ground station processors are networked via Ethernet links and are separately connected to the incoming telemetry data and command lines via RS-422 high speed serial interface boards. Each workstation is provided with a printer for local control of telemetry snaps, system messages, and status printouts. All incoming telemetry data are archived redundantly on digital 2.3 Gbyte 8-mm tape drives. Telemetry data is first buffered in ground processor memory and then subsequently stored off to either disk or tape.

As backup, QuickStar can provide its own Mission Unique Equipment (MUE) in the Consolidated Space Test Center (CSTC) of the Air Force Satellite Control Network (AFSCN) much like Delta Star and LOSAT-X (Figure 6-3). Data from the spacecraft is serially routed through the DSM to CSTC MUE where it is put on tape (still encrypted) and routed to the QuickStar ground station for processing and review.

The AFSCN can support QuickStar missions through the Consolidated Space Test Center (CSTC) and its worldwide network of Remote Tracking Stations (RTS). This has the benefit of providing QuickStar missions with satellite command and data coverage at selected passes on a 24-hour per day basis. A 9600 bps modem interface provides communication to and from the mission unique equipment (MUE) in the CSTC environment and also remote telemetry display capability to other non-colocated PCs. The MUE and software are identical to the command and control system

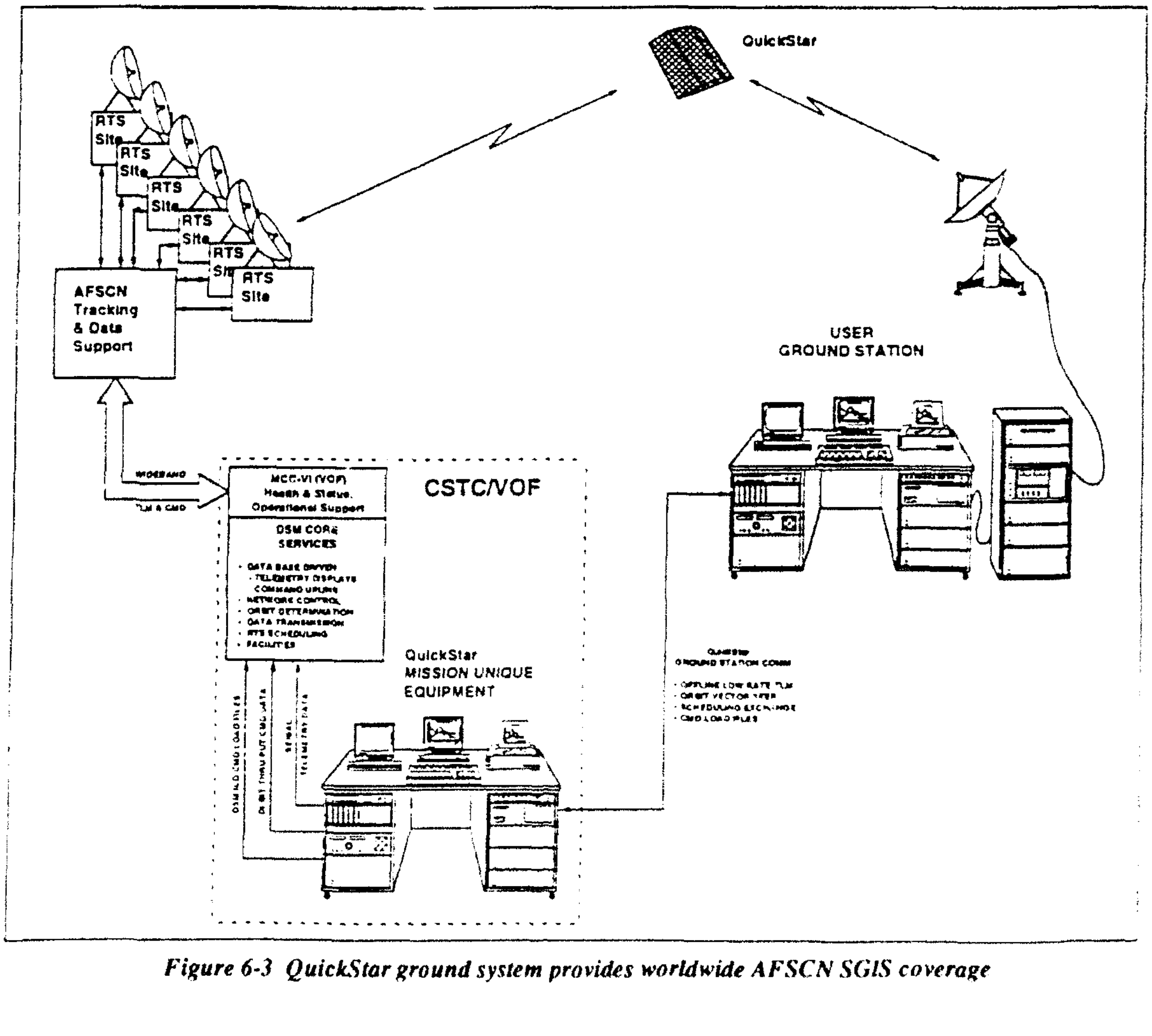


Figure 6-3 QuickStar ground system provides worldwide AFSCN SGLS coverage

used in the QuickStar ground station. The MUE receives all telemetry data from DSM via an RS-422 serial interface. Di-bit commands are generated in the MUE for throughput to the RTS network. The overall MUE approach was selected to preclude having to develop DSM related software, command, and telemetry data bases. Although, the approach still allows use of standard DSM services for network control, data interfaces, tracking, and orbit determination.

Orbit determination is performed at the QuickStar ground station. At the ground station, digital ranging data (without range rate) is collected and a single site solution is generated using the Microcosm 80386 based orbit determination system.

7.0 TACTICAL/PAYLOAD APPLICATIONS

The QuickStar configuration is tailored for small payloads of various configurations and its subsystems provide impressive capabilities to the tactical payload (**Figure 7-1**). Although QuickStar offers an optional propulsion capability, the weight of adding the propulsion system reduces the payload capability by approximately 15 pounds.

The QuickStar payload cavity has a volume of 1.7 cubic feet. The volume consists of three nearly equal cubes as shown in **Figure 7-2**. The bulkheads forming the cubes can be penetrated to form a single volume to accommodate larger instruments. In addition, 0.8 cubic feet of volume, located in front of the forward bulkhead, is also available for instruments longer than the payload cavity. However, the use of this volume is contingent on the particular primary payload being flown on the Delta II and must be negotiated on a case by case basis.

The capability to penetrate the bulkheads of the payload cavity allows a wide variety of instrument configurations to be integrated into the QuickStar. Various "real" instrument configurations being installed into the spacecraft are shown in **Figure 7-3**. Section A of the figure shows several small instruments being integrated into the payload cavity, with aperture penetrations for each sensor through either the forward bulkhead or through the solar array substrait. Section B shows how the internal bulkheads are penetrated to accept a long, small diameter instrument. Section C&D show other instrument and electronic module configurations that also can be integrated into the QuickStar.

CHARACTERISTIC	QUICKSTAR CAPABILITIES
Pointing stabilization	All-pointer (3-axis)
Pointing control	0.1 deg
Attitude knowledge	<0.1 deg
Slew rates	5 deg/sec (3-axis)
Payload power	40 W (orbit ave.)
Payload data rate	4 Mbyte/sec
Payload weight	70 lbm
Payload volume	1.7 cubic feet
Optional volume	0.8 cubic feet
On-board processing	2 MIPS
Data storage	1 Gbit
Downlink data rate	1 Mbits/sec

Figure 7-1 QuickStar payload capabilities

Small, smart systems can contribute to many tactical support missions in ways that are only beginning to be understood. Before these systems can become an integral part of the force structure, convincing mission analyses and utility assessment must be performed. A number of potential tactical applications for a small, smart space system are identified such as Earth observations/remote sensing, surveillance, communications, and technology demonstrations.

Battlefield Surveillance

Tactical commanders need to reliably and accurately detect, characterize, and target enemy forces. Advancing armor, surface-to-air (SAM) sites, airfields, supply depots, headquarters units, etc., all need to be located and delt with. A wide variety of battlefield conditions need to be monitored.

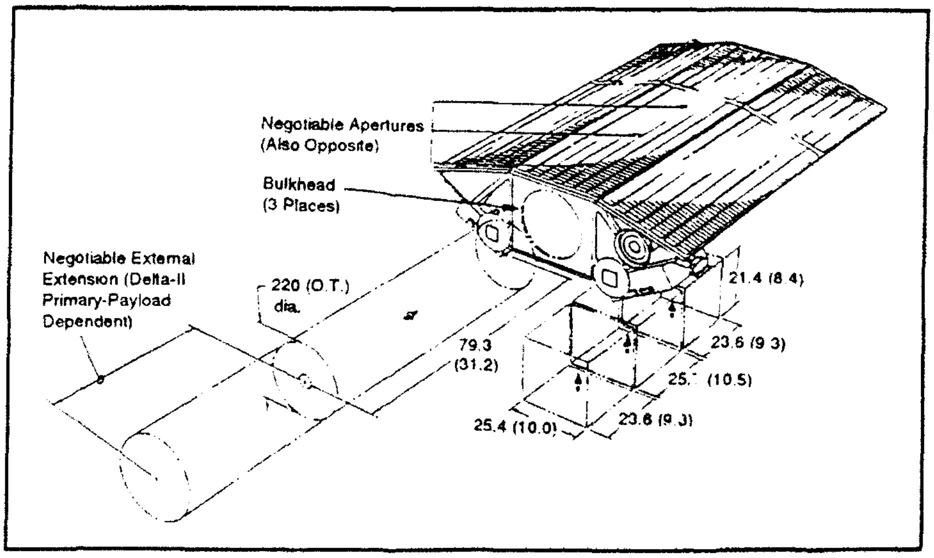


Figure 7-2 QuickStar payload volume

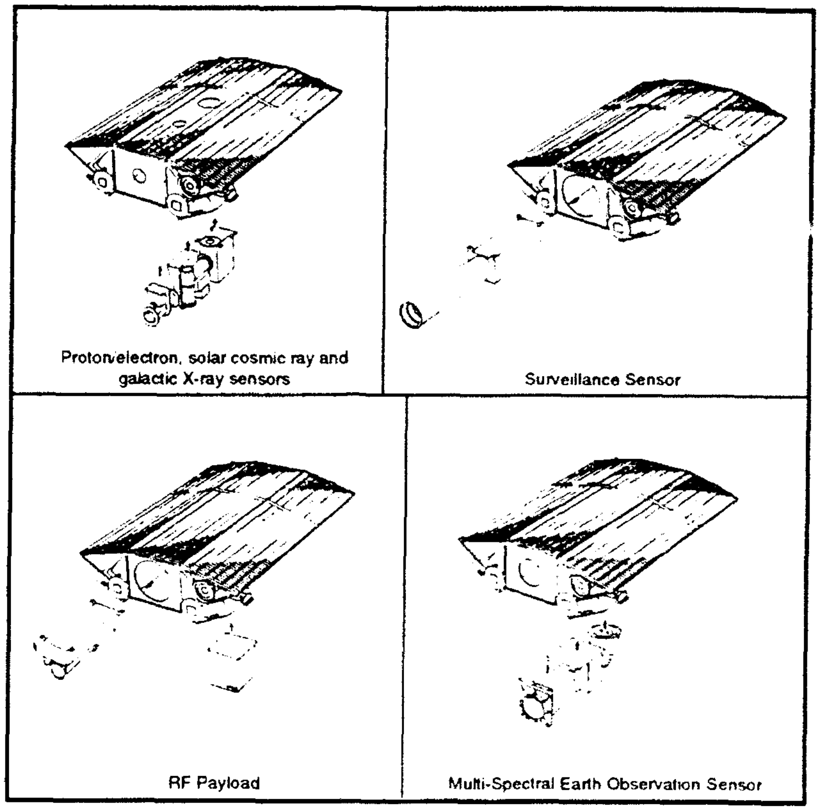


Figure 7-3 Four possible payloads that are compatible with QuickStar

Figure 7-4 depicts a small, smart space system carrying an electro-optical sensor. Reasonable size sensors at altitudes of 300 to 500 km can sweep selected regions of a battlefield with ground resolution of between 1.5 to 3.0 meters depending upon altitude and viewing geometry, sensor performance, size of the focal plane array, etc. Multi-spectral and day/night capabilities are also possible. A wide variety of targets can be detected, recognized, and even identified. Multi-spectral capabilities offer the potential for detecting camouflage, concealment, and deception usage by the enemy. Photon counting focal plane technologies can deny night-time cover. Direct wide-band data links to tactical vans can provide essentially real-time imagery for target identification and selection. Needless to say that while the scenario depicted in Figure 7-4 reflects a ground battle, many other scenarios are possible. As a directed search system, it can monitor harbors, naval bases, airfield, trainyards, and factories—again both day and night.

The capability to view a single site with a single satellite should not be the criteria upon which a space system should be judged. Viewed within the content of broad area surveillance in a 90-day war, the system could provide the information on thousands of sorties. A small, affordable space system would be both highly mission and cost effective. This is doubly true if combined with other tactical support missions such as tactical and weather monitoring.

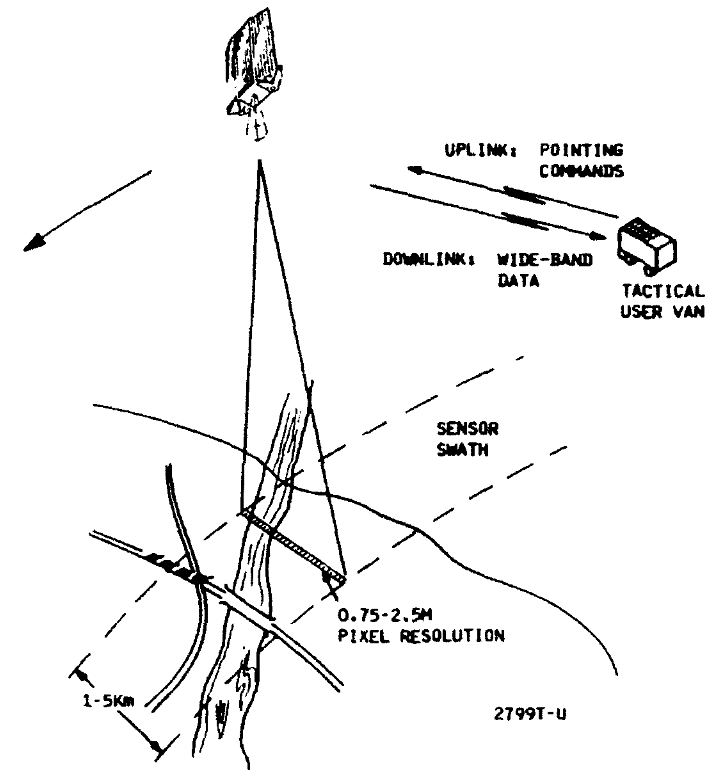
Tactical Signal Collection

Battlefield RF signal collection is similar to battlefield electro-optical surveillance: the payoff to the tactical commander is very high. Although the tactical commander has organic signal collection assets, they must normally be put at risk to be used and they suffer from range limitations. A space-borne directed search capability under the control of the tactical commander would complement the ground, sea, or airborne assets.

A typical mission scenario is shown in **Figure 7-5**. Consider the need for a strike against a deep airfield or supply area; space-borne assets can contribute. Small highly-capable spacecraft with high gain, narrow beamwidth antennas can be directed by the tactical commander to scan the desired areas slowly, dwelling on detected emitters if necessary. During the pass, the space-borne assets can view the desired areas for several angles and for periods up to 10 to 15 minutes. Detected signals could be processed on-board and relayed directly to the tactical users. Stored command and data capabilities would allow over-the-horizon operations.

Gap-Filling, Reconstitution, and Special Communication Missions

FLTSATCOM, DSCS III, and the future MILSTAR and UHF Follow-On Programs are capable, long-lived systems.



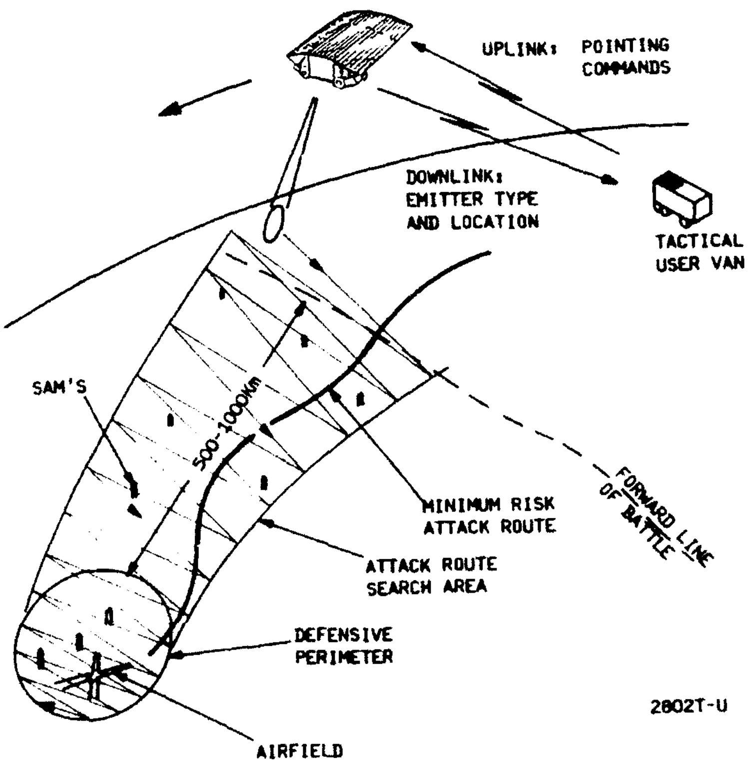
MISSION

- Tactical target detection, characterization, and targeting

CONCEPT

- Small satellites with multi-spectral sensors 1.5 to 3.0 ground resolution
- Real-time, tactical user control
- Wide-band downlink to tactical vans
- On-board storage for limited over the horizon operation

Figure 7-4 Electro-optical battlefield surveillance



MISSION

- Detect, identify, geo-location tactical emmiters

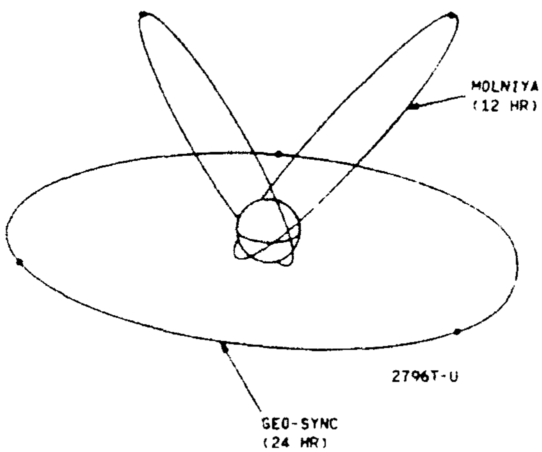
CONCEPT

- Small satellite with high gain steerable antenna
  - Stare and slow scan
  - Sidelobe intercept
- Tactical user tasking control
- On-board signal processing
- Direct down-link
- On-board storage for over the horizon operations

Figure 7-5 Tactical signit

Should any UHF, SHF, or EHF satellite be lost due to launch or on-orbit failure, the nation's ability to replace them quickly is severely limited. In short, we have essentially no capability to augment, gap-fill, or reconstitute our military communications capability.

Modest communication capabilities at UHF, SHF, and EHF can all be implemented on QuickStar/Pegasus/Scout class spacecraft (ref. Figure 9-1). These spacecraft can be launched by Pegasus or Scout boosters. With high performance orbital transfer systems they can achieve Molniya and Geo-synchronous orbits (Figure 7-6). Payloads with antenna/transponder weights of 60 to 200 lb are possible. Continuous electrical power of 40 to 100 W to the communications payload is available.



MISSION

- Augment, gap-fill, back-up DSCS or MILSTAR

CONCEPT

- Small, limited channel satellites launched on-demand
- GEO-synchronous and Molniya orbit options
- Compatible (transparent) to existing ground terminals
- 90 to 180 days autonomous operation
- Small ground control vans

TS.2.9

Figure 7-6 UHF/SHF/EHF communications

Reconstitutable Weather Support

The Joint Chiefs of Staff have defined needed meteorological support in terms of a number of critical issues:

- Timeliness: Delivery time from observation to user
- Refresh: Interval between observations of a given area
- Ocean fronts: Improved observations of fronts and eddies
- Cloud type: Distinguish cloud types

The current Defense Meteorological Support Program (DMSP) provides the primary weather support to the DOD. It is a capable system with direct downlinks to selected tactical users. However, since only two spacecraft are maintained on orbit, JCS guidelines for timeliness and refresh are not being satisfied. Also, the DMSP sensors are not adequate to support the ocean front observation requirements. The Navy currently relies upon a five-band Advanced Very High Resolution Radiometer (AVHRR) on NOAA's TIROS satellites. This data is key to many naval operations including anti-submarine warfare (ASW). The TIROS data is unencrypted and used by the Soviets; in time of conflict it will most likely be turned off. Finally, as a low altitude satellite, DMSP is subject to conventional laser, and nuclear ASAT attack. It is not rapidly reconstitutable. Its loss would severely impact the war fighting capabilities of nearly all tactical commanders.

Essential meteorological data can be obtained by a single instrument, specifically the five-band AVHRR. It is a relatively small instrument weighing approximately 75 pounds. It can be easily carried by small spacecraft and provide direct downlinks to tactical users. The concept is to add small satellites to the DMSP constellation improving the systems timeliness and refresh capabilities. These satellites would mimic the DMSP downlink thus ensuring interoperability with existing ground and sea-based receiving stations. Information vital to naval operations would be provided by military spacecraft with encrypted downlinks. Rapid reconstitution by small booster is both feasible and affordable.

Research and Development Support

Organizations that need to test experiments or prototype systems in orbit are faced with a problem. Either they can procure small satellites with limited capabilities for several million dollars or they must group missions together to be able to afford larger and much more costly spacecraft. Either a user paid $2M for an inadequate spacecraft or $40M for a complex, long lead-time spacecraft. The lack of an affordable middle ground has inhibited the use of space by the research and development (R&D) community.

QuickStar provides a low cost R&D platform that can accommodate a wide range of payload configuration, orbits, attitude stabilization requirements, power levels, and communication links. Much improved capabilities over current systems are possible using today's technologies.

8.0 FLIGHT HISTORY

On July 3, 1991, the prototype to QuickStar was successfully launched into low Earth orbit as a secondary payload aboard a McDonnell Douglas Delta II 7925 launch vehicle. The primary payload aboard the Delta II was the Air Force GPS-11 spacecraft. To minimize risk to the primary payload, the QuickStar prototype was completely inert (powered down) at liftoff, placed into orbit by the second stage and activated at separation, well after the GPS vehicle had been deployed.

At separation from the Delta, the QuickStar prototype powered-up all subsystems and payload, nulled body-axes rates, maneuvered to the required attitude, and approximately 34 minutes later performed a crucial element of the mission. Accomplishment of a mission event so soon after spacecraft separation represents atypical early orbit operation since most spacecraft require lengthy checkout periods before attempting any type of payload operations.

The low-cost SGLS compatible ground station and CSTC mission unique equipment operated as designed. The Boulder site supported 5 passes per day in addition to all engineering data analysis, mission planning and command generation.

9.0 COMPATIBILITY WITH OTHER LAUNCH SYSTEMS

The QuickStar system is also compatible with other small launch systems. **Figure 9-1** illustrates a reconfigured QuickStar spacecraft bus suitable for launching on the Scout or Pegasus launch vehicles. The new spacecraft bus configurations contain the same QuickStar flight-proven system elements referred to earlier in the paper. The Scout bus configuration is sized for multiple launchings: two copies of the new configuration with modest payloads fit within either the Pegasus or Scout fairings and are within their payload-to-orbit capabilities.

10.0 CONCLUSION

The QuickStar is not a concept—it is a flight-proven spacecraft design. The spacecraft and supporting subsystems including ground station can be delivered to a user within 15 to 18 months depending on user required modifications and payload interface complexities. The program schedule (**Figure 10-1**) for the design, fabrication, and test of the QuickStar satellite system reflects the fast paced environment of a low-cost program. Subsequent spacecraft could be delivered in 4 to 12 months if key component/subsystem purchases are initiated during the procurement of the first spacecraft.

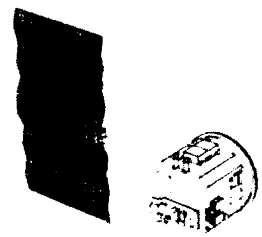
	QUICKSTAR	SCOUT CLASS	PEGASUS CLASS
Spacecraft Family Carrying Typical Payload			
Solar array	Fixed	Deployed / Articulated	Deployed / Articulated
Design life/goal (months)	6/12	6/12	12/36
Stabilization	Three-axis	Three-axis	Three-axis
Pointing accuracy/knowledge (deg)	0.1/0.1	0.1/0.1	0.1/0.1
Mass at separation (kg)	100	148	389
Array power (BOL, W)	214	542	588
Downlink rate, maximum (Kbps)	1,000	1,000	1,000
Payload mass, maximum (kg)	15 to 32	50 to 100	170
Payload volume, maximum (m3)	0.07	0.25	1.0
Payload power, orbit average (W)	25 to 50	50 to 100	100

Figure 9-1 A family of Smallsat buses using QuickStar subsystems

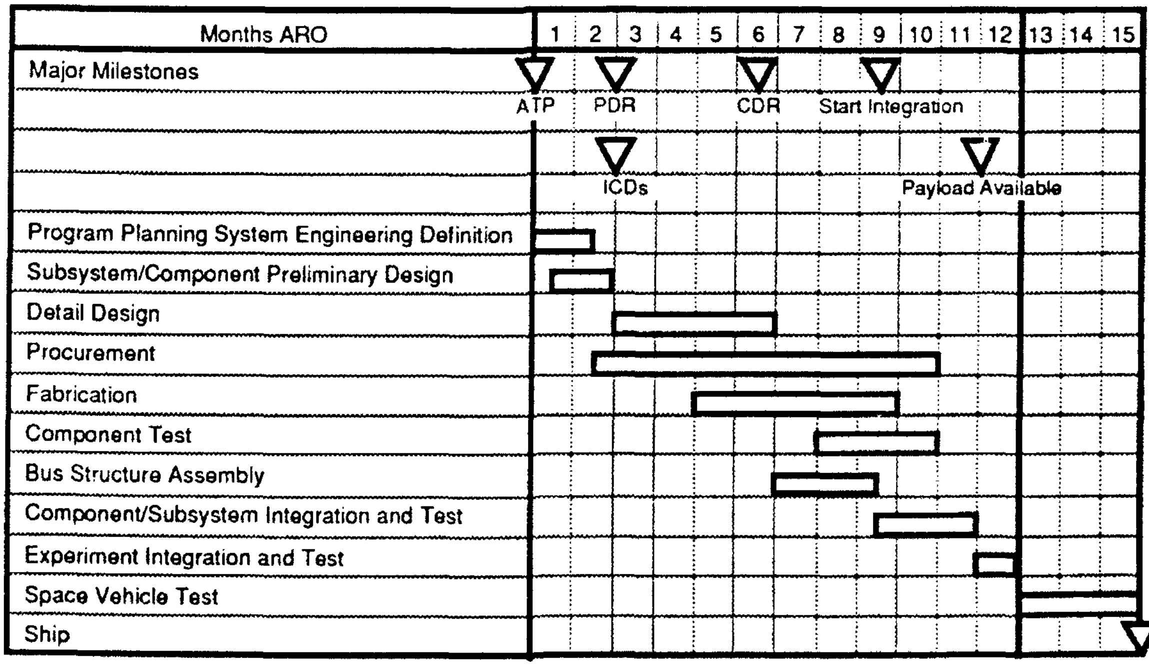


Figure 10-1 A fifteen month QuickStar satellite system program schedule

Attitude- and Orbit Control Subsystem Concepts for TACSATS

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1. INTRODUCTION

The technical performance specifications and requirements for Attitude- and Orbit Control Subsystems (AOCS) for satellites are dictated by a large number of boundary conditions:

- The payload requirements (e.g. pointing accuracy, LOS-stability)
- The perturbation environment, both external and internal, i.e.:
  - *Gravitational-, solar pressure-, air drag-, magnetic disturbance torques*
  - Reaction jets, wheel wobble, payload operation
- *Ease of operability* (command structure and sequences)
- Onboard autonomy (command-, data and communication links)
- Operational life time / reliability
- Maintainability, storabilty and so on

For the class of spacecraft under discussion here, mass limitations, which will not allow major orbit changes, and the different types of payloads for surveillance, verification and C3I as well as strategic aspects call for extremely high flexibility of the AOCS-concept to enable:

- "Last minute " selection of the operational orbits according to the actual need
- Adaptation to the requirements of different types of payloads (passive: Optical, infrared; active: μ-wave)
- Compatibility with a large dynamic range of plant parameters (e.g. large / small solar arrays and structural flexibility depending on electrical power requirement)

Modularity of the AOCS in terms of type and arrangement of equipment (sensors, actuators), control laws and operational sequences (OBC-capacity and algorithms) therefore has to be a predominant feature. It is expected that the flexibility for adaptation to specific mission requirements and the modularity of the AOC subsystem envisaged will contribute to a low cost but still highly efficient tactical satellite system.

On the basis of experience in Attitude- and Orbit Control System design, analysis, acceptance testing, and in-orbit operations support for communication satellites (INTELSAT V, TV-SAT/TDF, TELE-X, DFS-KOPERNIKUS, EUTELSAT II), earth observation satellites (MOS-1, ATMOS), and scientific satellites (ROSAT, EURECA, ASTRO-SPAS), suitable AOCS-concepts are discussed in the paper.

2. REQUIREMENTS AND CONSTRAINTS

In addition to services already generally available, tactical satellites of the type in question here are supposed to provide on short notice and particular request improved and efficient services in at least one, preferrably several of the following five mission areas:

- <u>communications</u>, e.g. from, to and between mobile units on ground or between satellites
- <u>weather</u>, i.e. detailed actual weather conditions and prediction in definite local areas

- missile warning, e.g. against tactical ballistic missiles, to possibly improve the efficiency of defensive actions
- navigation to contribute to the provision of highly accurate navigation data
- observation, surveillance and verification, e.g. to check disarmament treaty violation or - in a battle field scenario - to monitor force deployments and to provide target recognition, theater targeting and verification

As a consequence of the general mission objectives outlined above, operational and performance requirements, environmental conditions and constraints are imposed on the Attitude- and Orbit Control System, which are subsequently outlined to establish the general frame for the AOCS.

2.1 Mission and Payload Requirements

For various reasons like launch cost, mass in orbit, image resolution or power requirements of observation payloads the spacecraft under discussion will be injected into relatively low earth orbits (about 300 to 1000 km). In general near polar, sun-synchronous, cirular orbits are preferred, which are characterized by the condition that the product of three orbit parameters, i.e. orbit semi-major axis, exccentricity, and inclination has a specific numerical value, entraining the orbit node to rotate with one revolution per year. For optical observation paylaods "dawn orbits" are particularly suited. In this case the S/C will always view the surface of the earth at any given latitude at the same local time (see e.g. ref. 1, p. 68). A graphical representation of the orbit parameters "inclination" (in deg.), the "orbit period" (in minutes) and the "S/C velocity" (in m/sec) as a function of altitude (in km) for circular sun synchronous orbits is given in figs. 2.1-1a to 2.1-1c of Annex 1, respectively.

For such orbits the revisit period of specific terrestrial locations in equatorial regions or at low earth latitude for a single S/C may be unaccetably long for tactical earth observation requirements, whereas the revisit period in mostly uninteresting polar regions in principle equals the orbit revolution period. This situation can be improved as follows:

- For observation of a particular region on earth the S/C is launched into a dedicated orbit, the revolution period of which is an integer fraction of 24 hours. The revisit time will then be at least once or even several times a day. For instance (ref. figs. 2.1-1a and -1b) a satellite in sun-synchronous orbit of (about) 570 km altitude and 97.6 deg inclination will have an orbit period of 96 min and pass over the same area every 15th revolution. 16 orbits per 24 h (orbit period of 90 min) would even allow to revisit the same target every 12 hours (in the descending N/S and ascending S/N orbit crossing) but require an altitude of 275 km (inclination 96.56 deg) and the S/C would experience significant air drag as will be discussed later (fig. 2.1-3, Annex 1).
- A number of payloads can provide large ranges of "side looking" capability, enabling observation of a local area under different aspect angles in successive orbits. Except when phased array antennas are employed this generally requires mechanical slewing of optics, mirrors and/or reflectors and in turn may given rise to significant internal torques and associated attitude perturbations.
- Selection of low inclination orbits instead of quasi-polar orbits with inclination angles covering the range of geographical latitude of particular interest.
- Placing a sufficient number of satellites into appropriately inclined, mutually synchronized orbit to ensure the desired coverage probability.

The relationship between the number of satellites required to ensure continuous coverage from any point on the earth under a local elevation angle of at least 10 deg as a function of circular orbit altitude is shown in fig. 2.1-2 (ref. 2) of Annex 1. Proper selection of orbit inclination and phasing of the orbiting satellites is required. Particular examples of Low Earth Orbit (LEO) Communication Satellite Systems e.g. MOTOROLA/IRIDIUM or TRW-ODYSSEY requiring 77/12 satellites at 795/10350 km altitude are schematically indicated in this figure. The number of satellites decreases, of course if polar coverage is not required. In any case orbit inclination and phasing of individual S/C forming part of a satellite system have to be retained or respectively corrected in case of deviations from nominal during operational life time.

For the assessment of the amount (m_p) of monopropellant hydrazine required to maintain a circular orbit of given, relatively low altitude, reference is made to fig. 2.1-3 (Annex 1), where the drag factor (f) per day (d) and unit cross-selectional area (Q) of the S/C is plotted. The propellant mass is then given by

$m_p = f \cdot Q \cdot d$

When liquid bipropellant engines providing higher specific impulse are used, the propellant mass is about 25 % lower. In addition to altitude corrections inclination corrections may have to be performed. The inclination drift at altitudes between 280 km and 390 km amounts to about 0.2 deg/year and requires approximately 27 m/sec velocity increment for correction.

2.2 Spacecraft Physical Characteristics

It is, of course not the objective of this paper to elaborate on TACSAT mechanical, structural, power and payload design features. In view of the fact, however, that the AOCS is the S/C subsystem with the largest number of functional and hardware interfaces with other subsystems as well as satellite operational procedures, it is regarded necessary to establish at least a rough frame of the expected S/C physical characteristics:

- The S/C mass targeted for consideration in TACSAT missions is between 200 and 700 kg i.e. within the payload range of dedicated launchers like PEGASUS, TAURUS etc.
- Onboard electrical power generation capability of about 1 kW is regarded necessary to allow for
  - half orbit eclipse periods and
  - operation of microwave payloads (SAR typical average power 500 W, peak power about 10 kW/pulse)

  Consequently a solar array surface area of about 8 to 10 m^2 will be required corresponding to a mass contribution of about 16 to 20 kg if a typical weight factor of 2 kg/m^2 and use of conventional silicon solar cells are assumed. GaAs solar cells providing about 30 % increase in power output per unit surface area will have to be excluded for cost reasons.
- S/C geometrical configurations are generally determined by the constraints imposed by
  - the launcher cargo bay dimensions, shape and acceleration load,
  - the payload requirements (including power).

  If the fundamental configuration shall be retained for the different mission applications as outlined under section 2, the geometrical requirements associated with
  - optical observation payloads i.e. telescopes of sufficient focal length, operating in the visible and infrared region, and requiring cooling of the detector arrays (e.g. liquid helium dewers) and/or
  - microwave payloads (SAR) with large area (μ-wavefeed or phased array) antennas sufficiently extended in long track direction

  are expected to be the design drivers. Some square-shaped, modular central body truss structure with triangular, rectangular or hexagonal cross-sectional area seems to be principly adequate and will provide the necessary free surfaces for the housing the folded solar arrays and ensure unobstructed field of view for attitude measurement sensors and free space for attitude and orbit control reaction jets.
- Quite a number of existing desings (e.g. LANDSAT 6 - ref.3, MOS-1, see section 3.2) are equipped with a single wing solar array giving rise to large (solar) disturbance torques and disturbance torque changes during eclipse transitions, due to the unsymmetrical configuration. From the AOCS point of view a symmetrical configuration as generally adopted for geosynchronous communication satellites is of course highly desirable. In view of the required flexibility for free selection of the orbit inclination according to the tactical requirements the solar array is assumed to be orientable not only as usual around its axis of symmetry, but also about an axis perpendicular to the first axis of rotation. Motor driven deployment mechanisms as for instance implemented in EUTELSAT II can be easily modified to allow servo-controlled articulation of the solar array at the yoke junction to enable orientation of the solar array surface to the sun for a large range of orbit inclinations. Moreover by controlling the solar array rotation about two axis, solar pressure torque compensation techniques can be applied to improve attitude control performance and onboard angular momentum management.

In fig. 2.2-1a (Annex 1) such a variable solar array configuration capability is schematically indicated. Unsymmetrical deflection of opposite solar arrays w.r.t. the sun direction will generate solar pressure torques about an axis perpendicular to the sun incidence plane, counter-rotation as shown in Fig. 2.2-1b will give rise to "wind mill torques" acting around the sun line.

2.3 AOCS Operational and Performance Requirements

From the mission and payload requirements the detailed functional and performance requirements are derived.

Typical attitude control accuracy and line-of-sight stability requirements for the missions out-lined in section 2 are summarized in table 2.3-1.

Furthermore AOCS related Data Management and Control (DMC) tasks have to be performed, in particular:

- Surveillance and control of subsystems like thermal- and power monitoring and control
- Fault detection, isolation and recovery
- Mission- and operational telemetry and telecommand data handling

Operational tasks to be covered by the AOCS are for instance

- provision of complete onboard autonomy during periods of no ground station contact.
- orbit corrections on time tagged commands.
- updating and propagation of orbit model parameters and correlation with instantaneous attitude data.

3. ATTITUDE- AND ORBIT CONTROL SYSTEM CONCEPTS

Subsequently the characteristic features of conventional AOCS for communication- and application satellites will be outlined, alternative equipment and concepts for attitude measurement and control will be discussed and their feasibility for application in TACSAT missions will be assessed.

3.1 Typical Conventional AOCS Characteristics

AOCS technology and concepts of most S/C presently operational or in production are very similar. In particular the three-axis attitude stabilization principle is generally applied, primarily in view of the growth potential of the onboard power subsystem. For ease of reference the discussion of conventional AOCS characteristics to follow will be based on satellite families, for which the AOCS design authority was with DASA (Deutsche Aero-Space AG).

3.1.1 *Commercial communication satellites*

Within more than 20 years of AOCS design, manufacturing, qualification, testing and early in-orbit operation support activities for communication satellites, extensive experience has been gained in solving related control dynamics problems like stabilization of structural flexible modes, active damping of undamped oscillations via inertial coupling or handling varying instability conditions of sloshing propellant masses. In table 3.1-1 (Annex 2) a review of the S/C under discussion here, 24 of which are presently operational is given together with their launch dates. Table 3.1-2 (Annex 2) summarizes their specified AOCS performance figures and essential characteristics of the control concepts.

- The 3-axis stabilization technology for commercial geosynchronous communication satellites using the bias momentum principle was first established with the French-German experimental technology satellites SYMPHONIE. Both flight models operated successfully for more than 5 years. The S/C was spin stabilized in transfer orbit and during (liquid bipropellant) apogee boost and despun by yo-yo. Normal Mode (NM) attitude stabilization in GSO was performed with onboard closed loop pitch control using Infrared Earth Sensor (IRS) and fixed momentum wheel (FMW), roll/yaw corrections by cthruster pulses commanded from ground. Closed loop attitude control during orbit corrections was based on sun and (IR-) earth reference and modulation of 10 N liquid bipropellant thrusters.
- The series of 15 INTELSAT V S/C, developed, manufactured and launched in the period from 1976 to Jan. 89 was also spin stabilized in transfer orbit and during (solid propellant) apogee boost. Roll/yaw NM control is based on IRS (roll-) reference and the WHECON principle, using monopropellant hydrazine thrusters. Attitude control during orbit corrections was performed as established in SYMPHONIE with sun- and earth reference and thruster modulation. Two S/C were lost due to launch vehicle failures, all others are still operational, partly already exceeding their design life (7 years) by a factor of 2. Dedicated, automatic sun acquisition and safety modes and concepts for stabilizing solar panel structural flexibility effects have been developed.
- A significant progress in AOCS technology has been made during development of the direct television and broadcasting satellites TV-SAT/-TDF and TELE-X. Apart from new designs of attitude measurement equipment (sun-, earth

sensors, gyros) incorporated in the AOCS, the 3-axis stabilization technology for transfer orbit operations and during (repeated) apogee boost maneuvers has been established, using for the first time a unified liquid bipropellant propulsion system for attitude control, orbit control and Apogee Boost Maneuver (ABM). Furthermore these S/C are equipped with a coarse body control and an additional antenna fine pointing system based on RF-sensing and control of the TX-antenna beam orientation w.r.t. a ground beacon to an accuracy better than ± 0.025 deg each axis. Special reacquisition concepts in case of attitude loss or battery failure (recovery after eclipse) have been incorporated. All attitude measurement, data updating, formatting, monitoring, mode sequencing and control functions, except antenna (structural flexibility) stabilization have been implemented in a central, digital, internally redundant onboardcomputer. TV-SAT FM1 had to be deorbited because one solar array wing failed to deploy in GSO and simultaneously blocked deployment of an antenna reflector. The robustness of the AOCS design operating then under most abnormal conditions, was, however, unintentionally demonstrated, as well as its flexibility offered by in-orbit reprogramming of the onboard computer, when for different attempts of rescue maneuvers, additional control modes and sequences have been implemented.

- DFS-KOPERNIKUS a smaller class S/C family of different geometrical configuration ("rabbit-ear" antenna arrangement, liquid bipropellant tanks in ABM direction) is 3-axis stabilized in TO and GSO, the solar arrays being fully deployed already in transfer orbit and during (repeated) liquid bipropellant Apogee Boost Maneuvers. For this operational mode the AOCS design has to cope with panel oscillations and propellant sloshing phenomena in overlapping frequency bands, the sloshing dynamics experiencing pole-zero inversion during the maneuver. Concepts and provisions for in orbit gyro calibration have been incorporated to also ensure compatibility with midnight launch conditions.
- As compared to the previous communication satellite in the EUTELSAT II AOCS the following innovations have been incorporated:
  - The capability to acquire the earth any time of the day
  - A S/W safe mode for minimization of outage duration in GSO
  - An optimum Nutation and Angular Momentum Control concept to ensure higher yaw accuracy (as compared to WHECON) in presence of high disturbance torques (NM roll/-yaw control)
  - The capability to perform orbit corrections with yaw reference from gyro, i.e. any time per day (also in colinearity regions)
  - High pointing accuracy during station keeping maneuver (SKM) transients
  - In-flight recording of the thruster firing history for propellant budget monitoring

3.1.2 *Earth observation and scientific application satellites*

World wide a large variety of low earth orbit satellites for all types of terrestrial and environmental observation as well as scientific and research purposes have been designed, developed and launched within the national programs of the respective countries or by international agencies (e.g. LANDSAT, SEASAT, SPOT, ERS etc.). They are equipped with dedicated, optical, infrared, multispectral or microwave payloads, their AOC being tailored for the individual requirements. Subsequently a short review of the typical AOCS characteristics of this type of S/C will be given on the basis of satellite examples, the attitude and orbit control subsystems of which have been developed under DASA responsibility.

Table 3.1-3 gives a review of these satellites, their development period and launch schedule. Table 3.1-4 summarizes their attitude control concepts and performances characteristics

- The AOCS for MOS-1, the first Japanese Marine Observation Satellite was designed and developed up to and including the level of development model hardware implementation and closed loop subsystem functional- and performance testing in a joint German/Japanese development and training program. Engineering- and flight model manufacturing and acceptance testing has then been performed under responsibility of Mitsubishi Electric Cooporation. The S/C were launched into about 900 km circular sun synchronous orbits. Orbit correction capability (and back-up wheel unloading) is provided by a set of hydrazine thrusters. Normal mode control is performed by two momentum wheels in V-configuration. Wheel desaturation is nominally performed using magnetic torquers in quarter orbit cycling.

- ROSAT, developed within a German, UK, US scientific satellite program operates in a 570 km circular orbit, inclined by 53° w.r.t the equatorial plane. Attitude reference is established by high precision star sensors and a set of 4 integrating gyros (one skewed). A coarse sun sensor assembly with 4π FOV providing two axis attitude reference is used for initial and emergency sun acquisition. For the scientific mission two operational modes are foreseen:
  - In scanmode the S/C rotates about the sun line and the telescope mounted perpendicular to the axis of rotation performs within 6 months a complete sky survey for detection of new X-ray sources
  - In pointing mode the telescope is rotated to directions selected by the groundstation for dedicated source observation

  In both modes of operation inertial attitude reference is derived from star sensor measurements and star identifications with an onboard stored star catalogue. A set of 4 reaction wheels in skewed arrangement generates the necessary control torques. A three axis magnetometer and 3 magnetic torquers are used respectively for determination of the earth magnetic field vector and wheel unloading. Due to equipment failures emergency strategies for attitude generation from magnetometer measurements and an earth magnetic field model onboard of the S/C had to be developed and implemented.
- EURECA, the EUropean REtrievable CArrier, developed by DASA under ESA contract, shall be the platform for 5 different scientific missions, where the first mission (launch and return by shuttle) performs experiments under μ-g conditions. The AOCS (a cooperation between DASA, MATRA, GALILEO, LABEN) is able to perform orbit change maneuvers (boost-up, boost-down, inclination correction) sun and earth acquisitions with a hydrazine thruster system and is sun oriented in normal operation with a low level cold gas system supported by magnetic torquers.
- ASTRO-SPAS is a reusable satellite platform built by DASA under DARA contract, which shall fly once per year, beginning in 1993. It will be launched with the US Space Shuttle, set free in orbit and also berthed from the same shuttle. Its ACS is equipped with a rate integrating gyro package, a high precision star sensor, a GPS receiver and a set of cold gas thrusters. Similiar to the ROSAT mission two principle modes of operation can be performed:
  - In pointing mode the telescope is oriented into ground selected orientations
  - In scanmode earth atmosphere observations are performed where the earth reference information frame is obtained from the GPS receiver.

  As in ROSAT, in both modes inertial attitude is obtained from star sensor measurements and star catalogues.

3.2 AOCS Alternatives, Trade-offs and Trends

When discussing AOCS alternatives for TACSAT applications, the aspcets associated with attitude reference generation, force- and torque generation and attitude control concepts including the related equipment and software have to be adressed.

3.2.1 *Attitude reference generation equipment*

The general request for low cost solutions implies that attitude measurement equipment should be used, which just satisfies the subsystem performance needs. In view of the large variety of mission objectives and associated performance requirements (see table 2.3-1), however, equipment of equivalent measurement quality standards (and costs) has to be selected. Consequently the AOCS concept must provide the necessary flexibility to tie-in different kinds of measurement devices as requested by the application case. The types of equipment in question are of course Infra-Red earth Sensors (IRS), Precision Sun Sensors (PSS), STar-Sensors (STS) and Rate Integrating Gyros (RIG). Apart from conventional design performance characteristics, new designs are of particular interest. For the typical performance data, reference is made to table 3.2-1.

3.2.2 *Force and torque generation technology*

Spacecraft directly injected into target orbits with limited orbital accuracy demands and/or or relatively short operational life time may not require orbit corrections and consequently also no propulsion subsystem. In general, however, orbit corrections will be necessary to achieve and maintain the desired operational orbit. In view of aspcets like inherently different propellant utilization efficiency, the burn-out (or dry) mass and subsystem complexity, the most appropriate solution primarily depends on the total velocity increment to be generated for the mission in question.

In tables 3.2-2 and 3.2-3 characteristic parameters of catalytic, monopropellant hydrazine- and liquid bipropellant thrusters respectively are summarized. For ease of reference, again equipment available from DASA in-house manufacturing has been used as a basis of comparison. The parameters of the liquid bipropellant reaction jets refer to 'second generation' thrusters. They differ from previous and conventional design in that the combustion chamber is made of Platinum-Rhodium alloy, which allows higher operational temperatures and better efficiency than the classical regenerative chamber cooling principle. High pulse reproducibility is ensured by the swirl atomizer injection instead of cone injection principle used.

The diagrams of figures 3.2-1 and 3.2-2 show the specific impulse and impulse bit size respectively of the DASA 10N and 4N 2nd generation thrusters in pulsed mode of operation as function of thruster ON-time in comparison to conventional reaction jets. Small minimum impulse bit size (and accurate reproducibility) are particularly important features from the AOCS point of view.

Figure 3.2-3 shows the graphical representation of trade-off results, performed to identify the benefit of liquid bipropellant propulsion subsystem total mass including tanks, propellant, piping, and thruster system for a satellite with 8 small AOCS-thrusters as compared to mono-propellant technology. It turns out that the liquid bipropellant technology is alway superior and offers increasing mass benefit over monopropellant systems for increasing total impulse (Ns) to be generated.

While during orbit correction maneuvers the disturbance torques encountered during Δv-generation due to S/C centre of mass offsets are usually relatively high and necessitate counteraction by (the same) thruster system in ON- or OFF-modulated operation, external torques for attitude control or momentum management in normal operation may also be generated by magneto-torquers. The relatively high magnetic field strength of the earth at low orbit altitudes is particularly suited for magnetic torquing.

Table 3.2-4 summarizes the most important parameters of torque rods as manufactured by ITHACO. High performance attitude control and stabilization of all S/C axes in the necessary dynamic range is best accomplished by angular momentum storage devices i.e. reaction- and momentum wheels. Fig. 3.2-4 gives a small selection only of possible wheel arrangements and combinations for generating (internal) control torques about one or several S/C axes and/or bias momentum perpendicular to the orbit plane. As already mentioned with the typical AOCS concepts of section 3.1 the bias momentum principle offers particular advantages if attitude reference about all three S/C axis is not continuously available. Therefore momentum-/reaction wheel combinations are expected to cover most applications. A summary of characteristic parameters of off-the-shelf equipment (manufacturer: TELDIX) is given in table 3.2-4 of Annex 2.

3.2.3 *Feasibility assessment of control concepts for TACSAT missions*

From the previous discussions of mission objectives, control concepts, alternatives and equipment trade-offs and the AOCS requirements as summarized in table 2.3-1 it becomes obvious that

- the classical bias momentum system with two-axis attitude reference and WHECON control can only satisfy the needs for communication missions, where no particularly stringent requirements are imposed on the Line-of-Sight (LOS) stability.
- In the case of weather satellites, for the limited periods of scanning or picture-taking adequate LOS-stability must be ensured, absolute attitude pointing accuracy is not so critical
- For earth observation missions, depending on the observation payload concept (push-broom, detector array, scan mirror, multi-spectral decomposition) and ground resolution both accurate pointing and LOS-stability become increasingly important. Payload operation will have to be interrupted during orbit correction and transient periods. In normal mode wheel control about all three axes accompanied with high precision attitude reference has to be foreseen.
- Surveillance and verification being also earth observation tasks necessarily impose at least the same requirements on the AOCS partly even in presence of payload-induced perturbation environment. μ-wave payloads additionally require highly accurate orbit data (GPS), LOS stability and accurate attitude reconstitution over long periods (minutes). Payload data processing and payload/attitude data fusion is generally required entraining high demands on onboard computa-

tional and storage capacity and TM-priority during ground station contact.

4. AOCS BASELINE CONCEPT

In the past S/C design has often been performed with the emphasis placed primarily on mechanical and structural configuration aspects without taking into account the impact on other subsystems to the necessary extent. As far as the AOCS is concerned, it is of course understood that its S/C internal and external communication- and access capabilities can provide the necessary flexibility for adaptations to a given environment but disregarding certain demands from the AOCS on the S/C bus and equipment configuration would entrain the necessity to establish a dedicated AOCS-concept and make a special design for each mission objective and associated payload under discussion here. The approach outlined subsequently aims at making best possible use of the inherent flexibility features of the AOCS for the benefit of the overall system and is expected to meet the different mission requirements without major modifications. General features of the concept in question are:

- The AOCS shall be designed to incorporate sensor and actuator equipment alternatively or in combination as required to meet the pointing and LOS stability performance of table 2.3-1.
- The Data Management and Control (DMC) functions of section 2.3 shall be integrated with the AOCS (Integrated Control and Data Management System - ICDS).
- Data transmission within the S/C shall be performed by means of a digital serial data bus.
- H/W and S/W implementation shall be based.
- The capability of performing orbit corrections shall be provided.
- Payload data management (and preprocessing), which may necessitate extremely high computational and storage efforts is regarded as a task completely separated from the ICDS.

4.1 Sensor Configuration

It is understood that the whole scale of missions under discussion here can be best served, if the highest performance attitude measurement equipment is selected, i.e. precision star sensors and rate integrating gyros in strapdown mode of operation. This will, however, also be the most expensive solution and represent an overdesign for less ambitious, e.g. communication applications. Subsequently the attempt is made to define a sensor configuration, which could cover all situations, but inevitably imposes restrictions on the system lay-out.

It is assumed that TACSATS on request of strategic demands shall operated in between near polar (sun synchronous) down to near equatorial low earth orbits, the sun incidence angle with respect to the orbit normal also possibly varying between zero and 90°. If it is furthermore assumed that the array design provides the double axis orientation capability discussed earlier, an orientation of the solar array axis of rotation approximately parallel to the direction of motion is favourable (see fig. 4.1-1a)

- for very low altitude orbits (< 500 km) due to air drag (see fig. 2.1-3)
- if the sun is in a region around the normal to the orbit plane (say ± 45 deg)
- for low inclinations of the orbit plane w.r.t. the equator

An orientation of the S.A. axis of rotation perpendicular to the direction of S/C motion (parallel to the earth surface) is more favourable (see fig. 4.1-1b)

- for high altitude, high inclination (near polar) orbits
- for low sun incidence angles w.r.t. the orbit plane

The highest flexibility would be ensured, if - for respective application orbits - the S/C orientation w.r.t. flight direction could be selected either way and the payload (pointing in nadir direction) alternatively rotated by 90 degrees around its line of sight in the S/C.

A feasible arrangement of all (optical) attitude measurement equipment, is schematically indicated in fig. 4.1-2. The S/C axes are denoted by x, y, z (roll, pitch, yaw), where the yaw-axis is always nadir-pointing. The schematic of optical sensor arrangement on the S/C body surface of fig. 4.1-2 shall not postulate that all sensors indicated have to be available simultaneously in the same S/C. For communication missions e.g. the expensive sensor is not required, in high performance missions with stellar-inertial reference on the other hand the IRS is obsolete.

The conical scan IRS points into S/C y-direction (or opposite to it, depending on the sun incidence angle w.r.t. the orbit), which is the direction of motion in the case of fig. 4.1-1a and scans the earth underneath under a 45 deg cone angle. The star sensor is mounted on the -z face (away from the earth) inclined into the y-direction (or opposite to it) scanning the sky as the S/C rotates around its x-

axis during orbit motion. The sun is assumed to be around the x-axis in a cone up to about 70 deg (about 45 deg ± 23.5 deg inclination, polar orbit case). Two double axis fine digital sun sensors with ± 64 deg measurement range (both axes) are mounted in the S/C x/y plane with their optical axes inclined w.r.t. the x-axis by 60 deg.
In the case of fig. 4.1-1b the S/C is rotated around its z-axis by +90 deg, the x-axis then being the direction of motion, the IRS and STS turned away from the sun direction, which then may be close to the S/C orbit plane.

4.2 Actuator Configuration

Following actuators will be used

- 4N reaction jets for attitude control and orbit correction maneuvers
- Momentum wheels and reaction wheels for attitude control

The arrangement and the torque/force generation concept of the different actuators is described subsequently.

4.2.1 *Reaction Jets*

The attitude control and orbit correction maneuvers are performed by 12 double valve thrusters, see figure 4.2-1. The following notation for the different thruster sets is used for the subsequent discussion, where each individual thruster is denoted by a number and a character:

Set A := (1A, 2A, 3A, 4A)
Set B := (1B, 2B, 3B, 4B)
Set A/B := (1A, 2B, 3B, 4A)
Set B/A := (1B, 2A, 3A, 4B)
Set C := (1C, 2C, 3C, 4C)

Table 4.2-1 gives an overview of the thrust levels and the principal orientations of the different thruster sets.
Note that the 4 thrusters of set A or set B are placed and oriented symmetrically w.r.t. the COG and the principal axis, respectively. This arrangement implies, that the relation between the 4x1 thruster activation vector $\underline{a}$ and the 3x1 torque vector $\underline{\tau}$ applied on the S/C

$$\underline{\tau} = T_c\,\underline{a}$$

is qualitatively the same for these two thruster sets, or with other words, the 3x4 torque matrix T_c has always the form

$$T_c = \begin{bmatrix} t_{cx} & t_{cx} & -t_{cx} & -t_{cx} \\ t_{cy} & -t_{cy} & t_{cy} & -t_{cy} \\ t_{cz} & -t_{cz} & -t_{cz} & t_{cz} \end{bmatrix}$$

where t_{cx}, t_{cy}, t_{cz} are the nonzero absolute values of the torque levels around the S/C x-, y-, z-axis. They will be optimized according to system requirements (e.g. expected disturbance torque), by an appropriate choice of the placement, i.e. the lever arm w.r.t. COG and orientation of the thrusters.
For ideally mounted thrusters even the numerical values of t_{cx}, t_{cy}, t_{cz} are identical, due to the symmetry property. This fact has the consequence that in case of failure of one thruster of set A (or set B, resp.), the double valve thruster configuration permits to choose the thruster of set B (or set A, resp.) with the same number to assure the redundancy. Alternatively it can be switched directly to the healthy thruster set B (or set A, resp.).

Torque Generation
The relation between the torque vector $\underline{\tau}$ and the thruster activation vector $\underline{a}$ is requested to be

$$\mathrm{diag}\,(t_{cx},t_{cy},t_{cz})\,\underline{m} = T_c\,\underline{a}$$

where $\underline{m}$ is the modulator output vector with its elements consisting of -1,0,+1, i.e. there are 27 values of $\underline{m}$.
Using the singular value decomposition of the torque matrix

$$T_c = V\,\Sigma \begin{bmatrix} U_1^T \\ U_2^T \end{bmatrix}$$

it follows

$$\underline{a} = T_c^{+}\,\mathrm{diag}\,(t_{cx},t_{cy},t_{cz})\,\underline{m} + U_2\,c$$

with the pseudoinverse T_c^{+}. The constant c will be chosen such that all elements of $\underline{a}$ are ≥ 0.
The solution of this equation for all 27 combinations of the modulator output signal results in the values of 0,0.5,1 for the elements of $\underline{a}$. The value 0.5 will be realized "on the average" by operating the corresponding thruster over half the modulator output time increment.
Of course, the above equation will not be solved on board, but the solutions, i.e. the correspondances

between the modulator output m and the thruster activation a are stored in the on board computer memory according to table 4.2-2.

Force Generation

According to figures 4.1-1, there are two different S/C orientations in orbit. For both cases, table 4.2-3 summarizes the nominal and redundant thruster sets, which are used for orbit corrections in normal and tangential direction.

4.2.2 *Momentum and reaction wheels*

The preferred wheel arrangement as shown in fig. 4.2-2 consists of

- Two flywheels in a V-configuration in a plane rotated about the y-axis by an angle η
- Two reaction wheels along the x- and z-axes

In the nominal configuration only one flywheel (either FMW 1 or FMW 2) and both reaction wheels are in operation, all running at bias speeds such that a residual bias angular momentum vector is nominally perpendicular to the orbit plane.

In case of a failure of one reaction wheel the residual reaction wheel and both flywheels will be used.

If on the other hand the "nominal" flywheel fails, the cold redundant fixed momentum wheel is activated.

4.3 AOCS H/W and S/W Implementation

Presently, based on a cooperation agreement between DASA, Aerospatiale, and Alenia Spazio joint efforts are being undertaken to develop and qualify an "Integrated Control and Data Management System" (ICDS) for next generation communication and application satellites, within the so-called "Spacebus Improvement Program" (SIP). This program also incorporates development and qualification of advanced components like the precision sun sensor (table 3.2-1), second generation liquid bipropellant thrusters (table 3.2-3) and in particular also a powerful Onboard Computer Unit (OBCU) and the associated operational and application S/W. The ICDS in question (see ref. 10) is to the largest extent directly suited or easily adaptable for the case under discussion here. The main difference to the concept favoured for communication S/C applications consists in that for TACSAT missions a "two computer solution" instead of a one central compute approach is regarded absolutely necessary in view of the flexibility- and computational requirements of the different payloads. The amount of payload data to be transmitted to ground within short ground contact intervals will furthermore necessitate direct priority link to the telemetry system. Subsequently only the AOCS part of the ICDS concept, which then can be regarded largely independent from the payload data management system will be shortly outlined.

Hardware

Fig. 4.3-1 shows the ICDS hardware configuration (without redundancies) i.e. not only the AOCS-, but also the DMC hardware units are represented.

The functional sharing of these units and comments on the notations within figure 4.3-1 are listed in table 4.3-1.

The serial OBDH Data Bus is the link within a modular expandable AOCS.

As far as newly developed equipment and signal conditioning electronics are concerned (e.g. sun sensor electronics, UPSE), the interfaces are designed such as to directly match the data bus I/F requirements. The signal conditioning for classical, off-the-shelf equipment (earth sensors, gyros, wheels, star sensors) to the OBDH data bus format is performed in the Platform Interface Unit (PFIU).

The central component of the ICDS hardware is the On Board Computer Unit (OBCU). It has functional interfaces with the ground segment and all on-board S/C subsystems for the distribution of commands and the acquisition of data. Figure 4.3-2 shows the blockdiagram of the OBCU. The functional units are:

- Processor Module (PM)
  - 16 bit processor, MIL-STD 1750 A instruction set architecture
  - processing power at least 1000 kips for DAIS MIX (Digital Aeronautics Instr. Set - 20MHz)
  - RAM 128 kwords of 16 bits
  - ROM 96 kwords of 16 bits
- Telecommand Decoder Module (TC)
- Telemetry Module (TM)
- Reconfiguration Module (RM)
  - supports autonomous satellite failure detection, isolation and recovery (FDIR)
  - functional elements: Alarm Level Management, OBCU Reconfiguration Commands, OBDH-bus Reconfiguration Commands, Thruster On-Time Control
- Safeguard Memory (SGM)

- contains the H/W system configuration data and dynamic S/W parameters
- Precision Clock Module (PCLK)
- Power Converter Modules (PCV)
  - provides necessary voltages for the OBCU modules
  - provides central on board reference time counter

Software

The complete AOCS and related DMC-software (S/W) will be executed in the redundant ICDS-on board computer and runs fully automatic and autonomously under normal circumstances.
Before the mission, the S/W will be (pre-) stored in the PROM and copied into the RAM after its first activation in the orbit. This allows modifications - as far as the application S/W is concerned - at any time via telecommands ("memory load"). It is also possible to adapt important system tables and the S/W configuration from ground.
Telemetry and telecommand S/W parts are implemented according to the ESA "standard packet TM/TC".
A central role within the on board S/W plays the MOSES II operating system, an updated version of the MOSES operating system used in the shuttle pallet satellite (SPAS)-program. It is especially designed for the

- management of parallel processes and asychronous events under real-time conditions

but can also meet the requirements of

- strictly synchronous operation modes, e.g. sensor data acquisition, activation of the UPS.

The maintainability and adaption capability of the S/W-system necessitate a transparent modular functional structure, hierarchically organized and properly defined interfaces not only within the operation system, but also to the application S/W modules.
The application S/W-parts (jobs) are divided into one or several tasks. Additionally, there are specific error tasks, which can be attached to a job/task. They will be activated automatically by the operating system in case of a failure (exception handling).

4.4 Attitude- and Orbit Control Loops

4.4.1 *Acquisition control*

Because of the short visibility periods of low earth orbiting S/C from individual ground stations onboard autonomy of fundamental operational functions is extremely important. Automatic acquisition, reacquisition and safety procedures ensuring initial orientation and S/C survival (power & thermal conditioning) in case of attitude loss are such fundamental functions. Classical, well proven concepts consisting in a reorientation of a preferred S/C axis (and the active surfaces of the solar arrays) to the sun are also applicable here and will therefore not be discussed in detail (see e.g. refs. 4 to 10).

4.4.2 *On-orbit control loops*

In classical satellite attitude and orbit control concepts clear distinction is made between attitude control during orbit correction maneuvers and in normal mode (NM), these modes of operation employing essentially different control principles (e.g. reaction jet control and NM wheel control). For the ICDS concept under discussion here a common control approach is regarded superior.
The most obvious advantages are:

- Inherent back-up capabilities using different types of actuators if required.
- Modular 3-axis attitude measurement estimation and reference generation principle throughout.
- Smooth transitions between different operational conditions due to continuous state propagation of estimation and control variables.
- Activation of combined actuators e.g. for feedforward compensation in case of heavy predictable (payload) disturbances

A schematic blockdiagram of the on-station attitude control mode is shown in fig. 4.4-1.
The raw sensor data from sun sensors and earth sensors (in case of low accuracy performance missions) or star sensors and rate integrating gyros for high performance missions are processed to generate attitude information (ϕ^*, θ^*, ψ^*). Based on system models orbit parameter updates from ground and additional measurements (wheel speed), optimal estimates of system state variables H_x (prop. ψ), H_z (prop. ϕ), θ, ω_x, ω_z, (ω_y prop. ω_0), H_x, H_z and θ are generated. In application cases, where attitude reference is generated from earth and sun sensors only, during eclipse regions yaw attitude is propagated by means of the observer equations of the angular momentum in orbit coordinates suppressing the nutation by adequate filtering.
Under normal mode conditions attitude stabilization is performed by controlling the fly-wheel and reaction wheels in torque mode. In order to com-

pensate for the impact of limited resolution in the torque commands as well as unknown internal wheel friction torque, contribution of cross-axis angular momentum components and external disturbance torques under eclipse conditions, a model following loop is incorporated to match the wheel control torques to the corrrect values. This technique as used for closed loop AOCS dynamic bench testing with real hardware at DASA is well proven. In view of the fact that in the wheel control concept in question here the wheel speed variation exceeds the usual range considerably, a nonlinear friction torque compensation loop is additionally superimposed. In order to improve the pointing accuracy and stability to the level required, disturbance torque estimation and compensation is applied. These estimates furthermore ensure, that the necessary yaw attitude prediction accuracy during eclipse passage is achieved.

Stellar-inertial attitude reference generation when using rate integrating gyroscopse in strap-down mode and updates from gyro drift computed by onboard Kalman filter using observed star transit times has to rely on up-to-date satellite ephemeris and reference star catalog data stored in the onboard computer. These data have to be periodically uplinked together with corrections of the onboard time reference. A typical example of a star identification pattern over a period of 12 min. as for instance encountered during ROSAT mission in scan mode is shown in fig. 4.4-2. In case of large payload-induced disturbance torques feed-forward control has to be applied to ensure the necessary pointing stability for high performance missions.

4.5 Preliminary AOCS Mass and Power Budget

For the AOCS concept and implementation approach outlined above preliminary mass and power budgets for the individual H/W-items have been summarized in table 4.5-1. Assuming that for a particular mission not all but only the necessary equipment will simultaneously be integrated into the S/C (at principly the locations indicated in fig. 4.1-2) the total AOCS budgets for respective application cases can be composed.

For typical mass of a propulsion system depending on the total impulse to be provided reference is made to figure 3.2-2. The burn-out mass including 12 four Newton bipropellant thrusters, tank, piping, residual Helium in a blow-down system amounts to approximately 18.5 kg.

5. CONCLUSION

On the basis of more than 20 years of experience in AOCS design and development for communication- and application S/C in the paper under discussion here the attempt has been made to identify the configuration, the equipment and technology of an AOCS, which can provide the necessary flexibility for serving the large variety of TACSAT mission objectives and the associated performance requirements.

6. LIST OF REFERENCES

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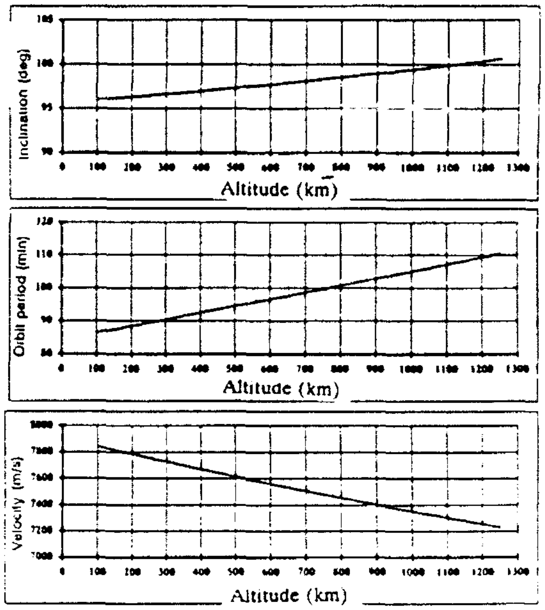


Figs. 2.1-1a-2.1-1c:
Parameters for sun-synchronous orbits

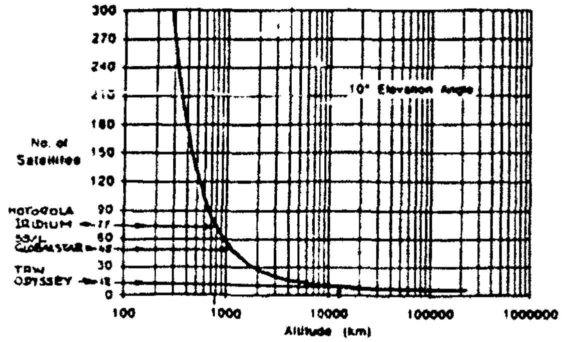


Fig. 2.1-2: Number of LEO satellites required for global coverage (10° elevation from ground)

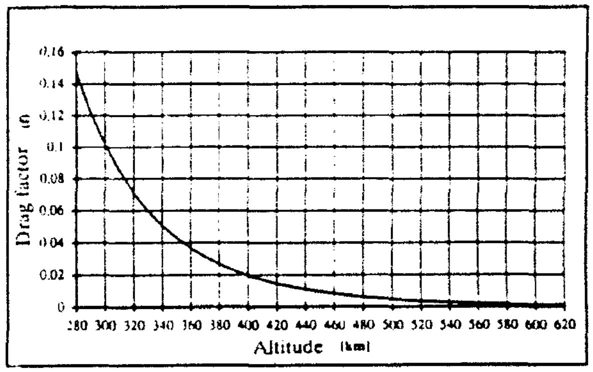
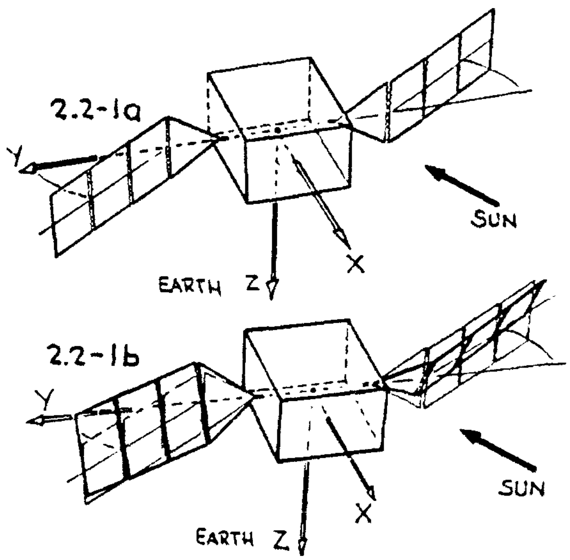


Fig. 2.1-3: Drag factor versus altitude



Figs. 2.2-1a/b: Principle S/C-solar array configuration

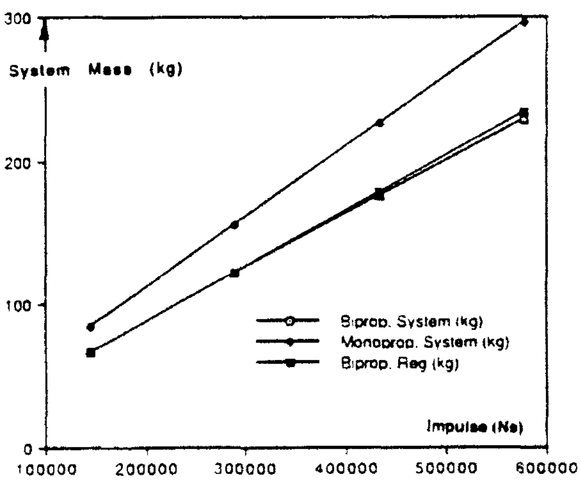


Fig. 3.2-3: Trade-off mono-/bipropellant thruster efficiency versus total impulse

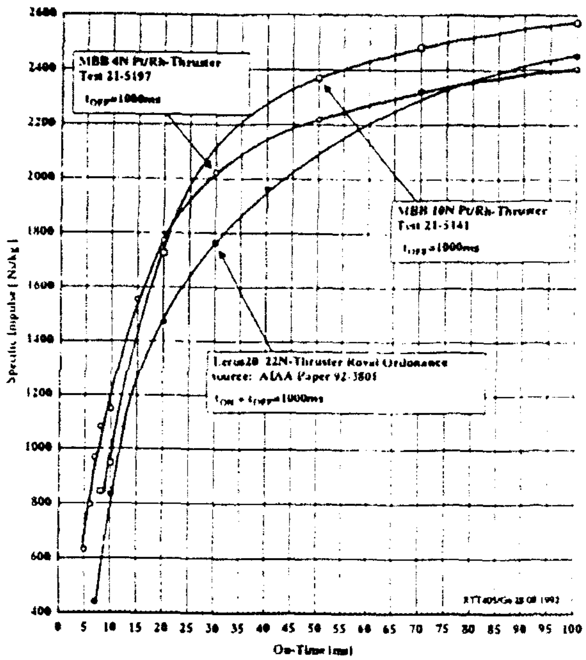


Fig. 3.2-1: Specific impulse of 2nd generation liquid bipropellant thrusters in pulsed mode (comparison)

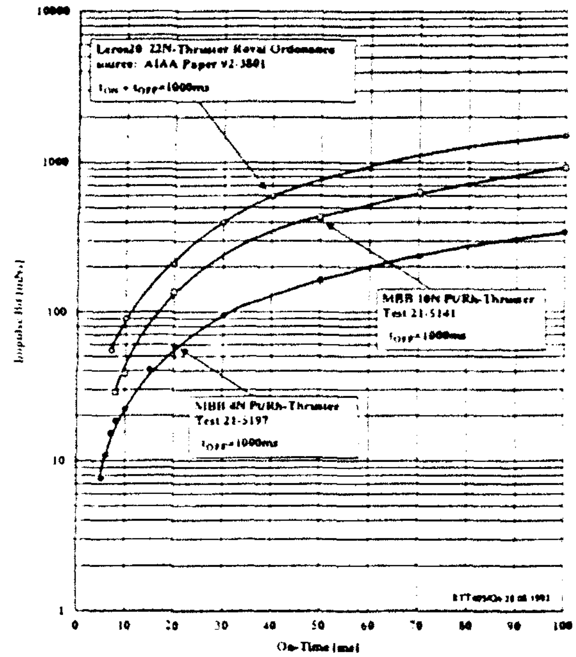


Fig. 3.2-2: Impulse bit of 2nd generation liquid bipropellant thrusters in pulsed mode (comparison)

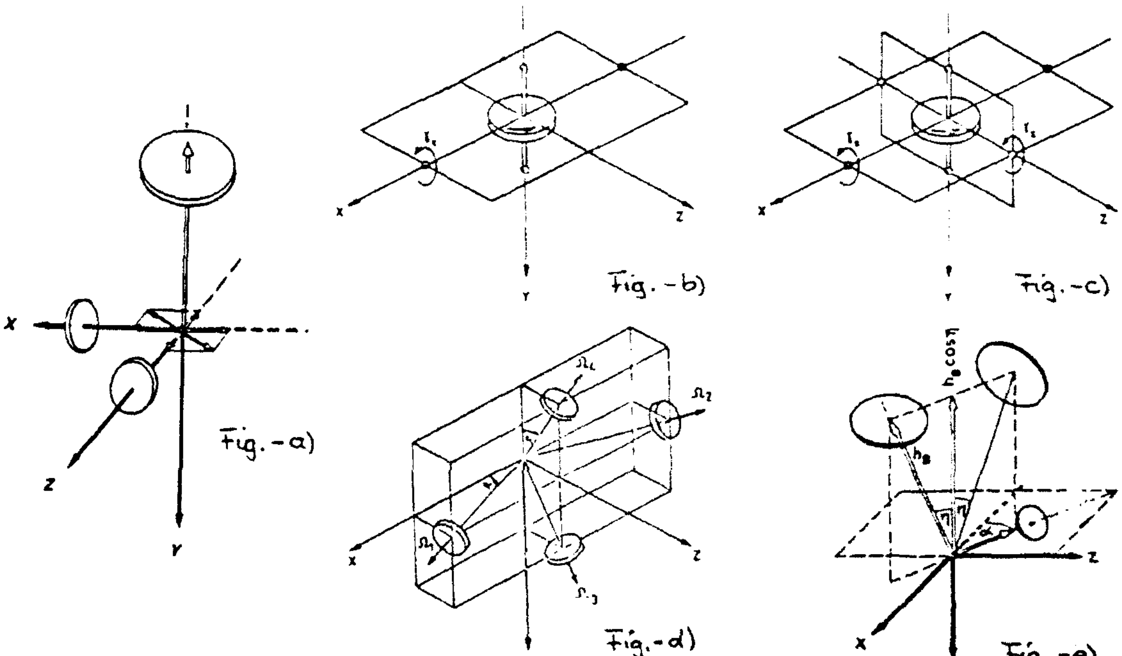


Fig. 3.2-4: Examples of momentum-/reaction wheel arrangements

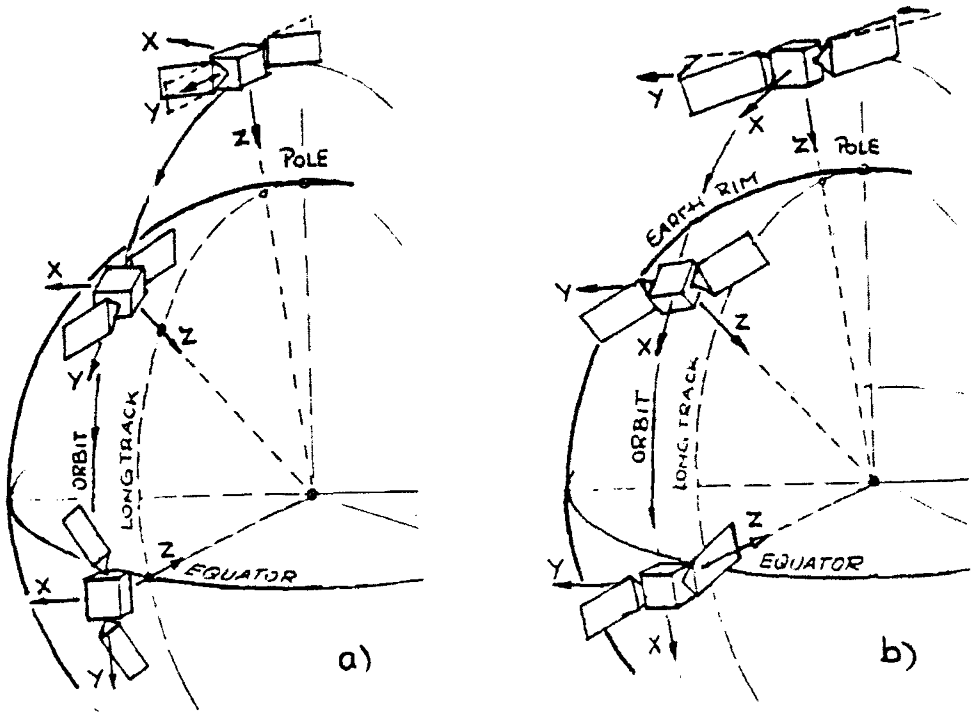


Fig. 4.1-1 a/b: S/C orientation 1/2 in orbit

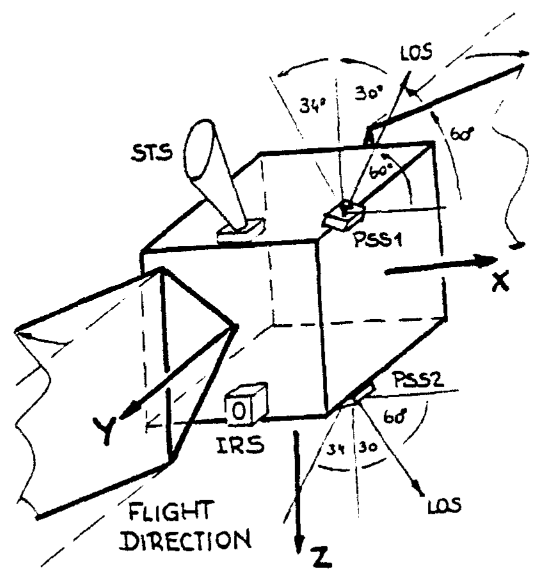


Fig. 4.1-2: Schematic of optical sensor arrangement

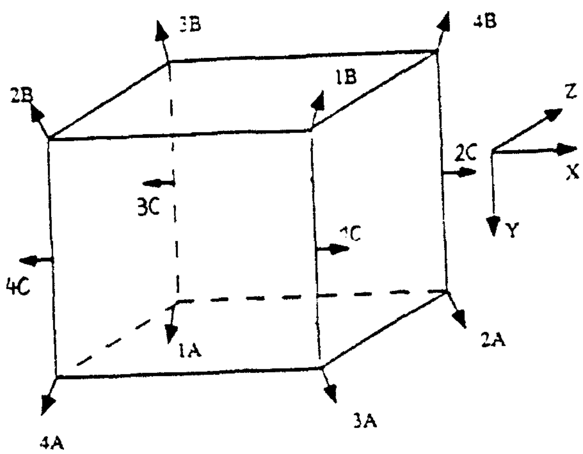


Fig. 4.2-1: Location and orientation of reaction jets

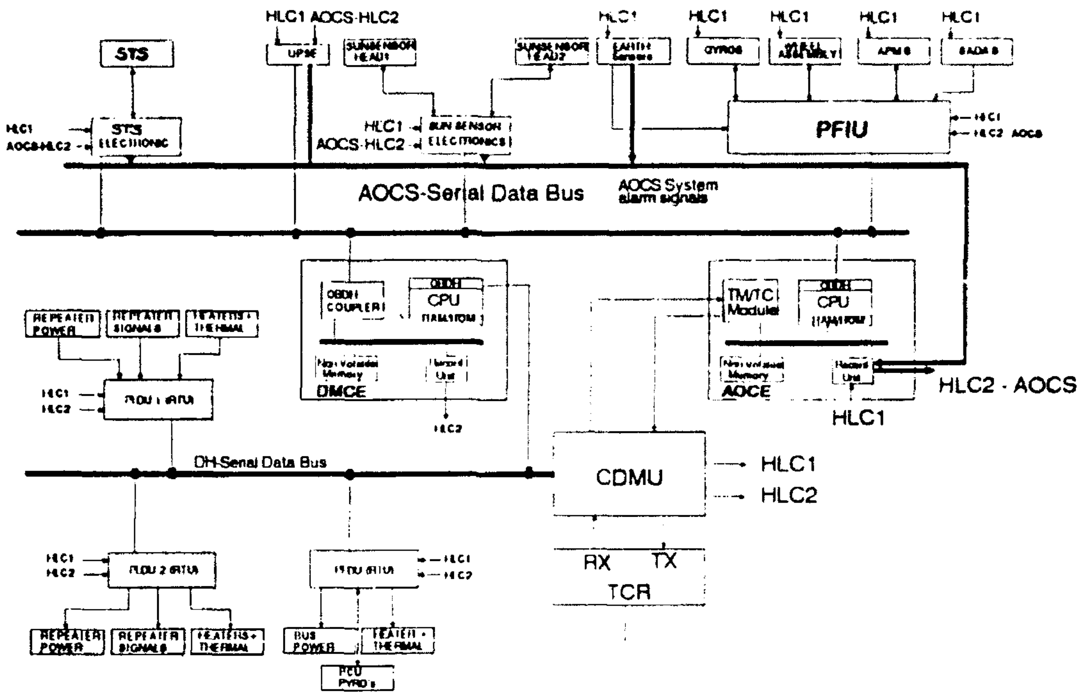


Fig. 4.3-1: ICDS principle block diagram for separated AOCS and paylaod DM

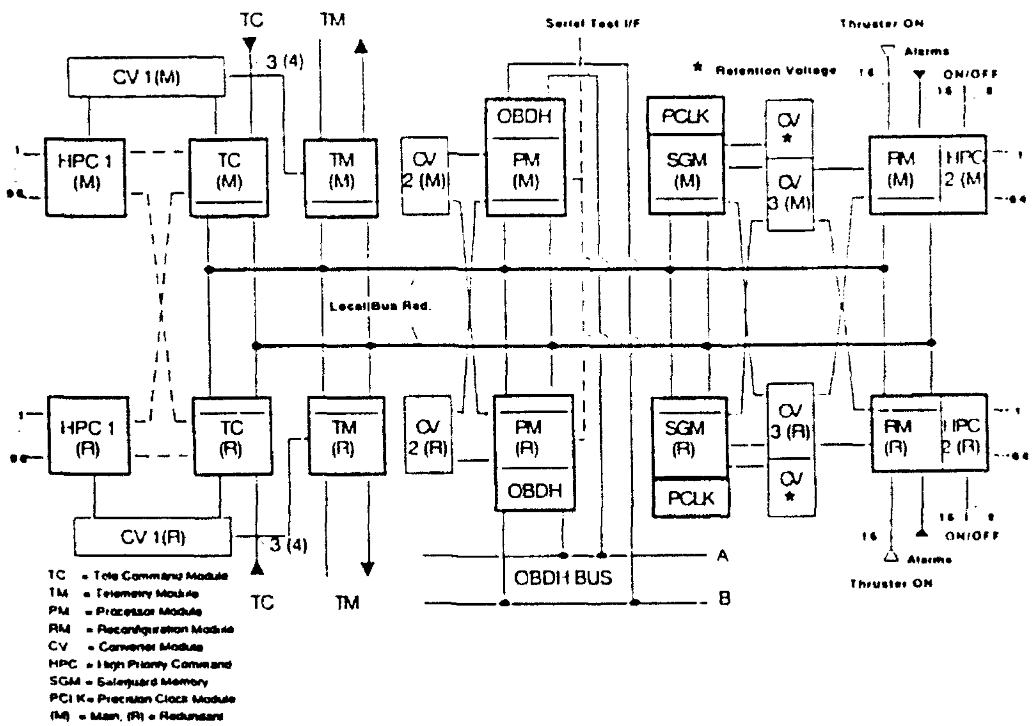


Fig. 4.3-2: Block diagram of the OBCU

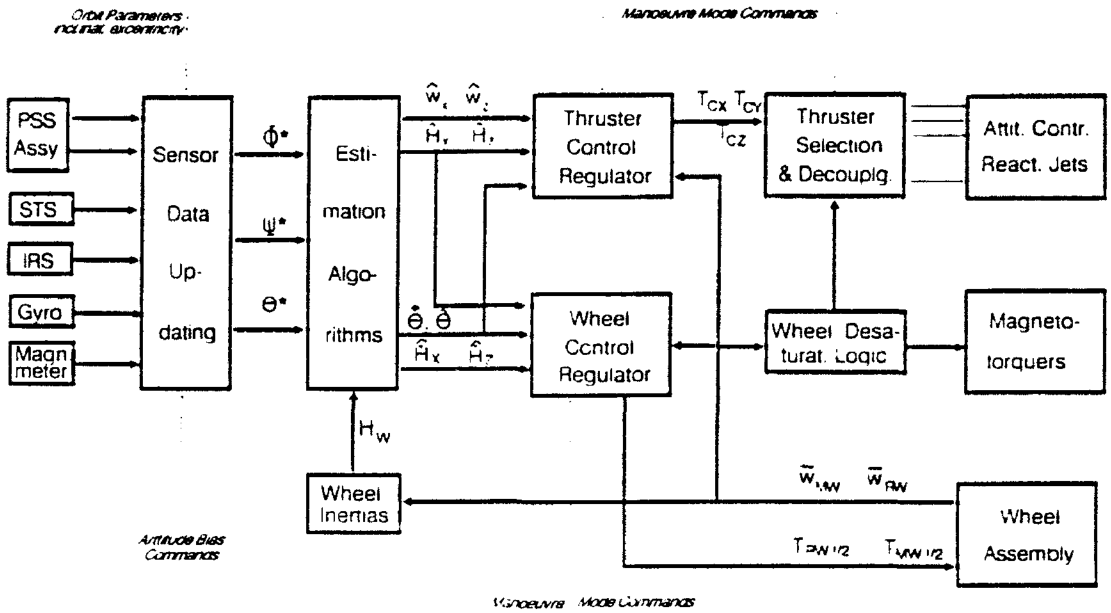
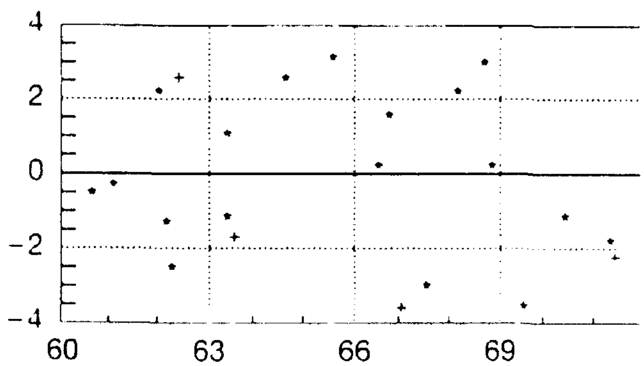


Fig. 4.4-1: Schematic block diagram of on-orbit control loop



star identification in scan mode
* stars in on-board catalogue
+ stars not in on-board catalogue

Fig. 4.4-2:

Star identification pattern in (ROSAT) scan mode

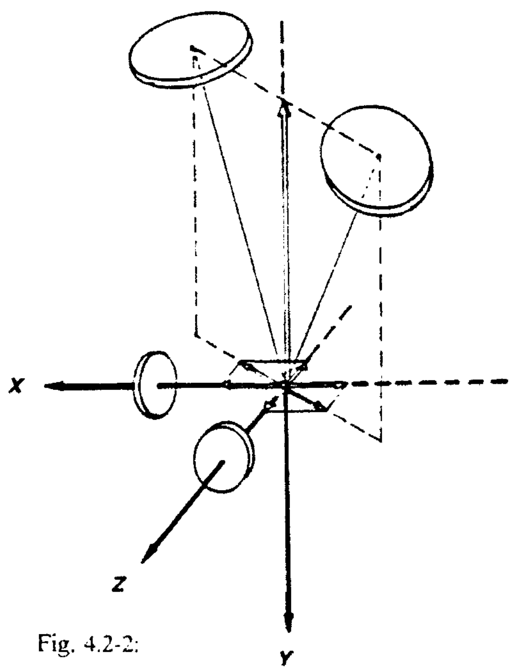


Fig. 4.2-2:

Momentum- and reaction wheel arrangement

Mission	Operational Mode	Pointing accuracy [deg]		Required Rate Accuracy [deg/sec]		
		half cone Roll/Pitch	Yaw	x-axis	y-axis	z-axis
Communication	Operational Phase	0.03	0.10	--	--	--
	Orbit Correction	0.15	0.35	--	--	--
Weather	Scan Phase	0.03	0.10	0.002	0.002	0.002
	Normal and SK-Mode	0.10	0.30	--	--	--
Environmental Monitoring	Normal Mode	0.03	0.05	0.002	0.002	0.002
	Re-Positioning	0.05	0.15	--	--	--
High Resolution	Normal Mode	0.007	0.02	0.0005	0.0005	0.0005
	Re-Positioning	0.05	0.15	--	--	--

Table 2.3-1: Typical AOCS performance requirements

Geostationary Communication Satellites		Devel. AOCS FM Qual.	Launch Dates
SMYPHONIE	2 flight models	1968 - 1970	A: 19.12.84, B: 27.08.75
INTELSAT V	15 flight models 13 Satellites in operation, FM 9 & FM 14 lost due to launcher failure	1976 - 1987	FM1 : 7.12.90 FM2 : 24.05.81 FM3 : 15.12.91 . . FM15: 27.01.89
TV-SAT 1, 2	2 flight models FM 1: one SG failed to deploy, deorbited	1980 - 1987	FM1 : 21.11.87 FM2 : 08.08.90
TDF 1, 2	2 flight models	1980 - 1987	FM1 : 28.10.88 FM2 : 17.07.90
TELE-X	1 flight model	1987	FM1 : 29.03.89
DFS-KOPERNIKUS	protoflight model FM2, FM3	1983 - 1989	PFM : 05.06.89 FM2 : 17 07.90 FM3 : 12.10.92
EUTELSAT 2	PFM = FM1 FM2, FM3, FM4, FM5.	1986 -	FM1 : 31.08.90 FM2 : 16.01.91 FM3 : Dec.91 FM4 : 10.07.92

Table 3.1-1: Review of geostationary communication satellite examples

	Control Accuracy		Control Concepts
	Half Cone [deg]	Yaw [deg]	
INTELSAT V	0.12	0.52	- Wheel control in Pitch NM - Whecon control in Roll/Yaw (fixed bias momentum wheel) - Spin-stabilized apogee boost maneuver
TV-SAT, TDF TELE-X	0.22 Beam Pointing Tx <-- Antenna --> Rx 0.035	1.2 0.07	- Wheel control in Pitch (NM) - Coarse body control in Roll/Yaw (Whecon with fixed bias momentum wheel) - Fine antenna pointing system using RF-sensors - Three axis stabilized apogee boost maneuver with partially deployed solar generators
DFS	0.07	0.3	- Wheel control in pitch - Whecon body control in Roll/Yaw (fixed bias momentum wheel) - three-axis stabilized apogee boost maneuver with fully deployed solar generators
EUTELSAT II	0.05	0.15	- Wheel control in pitch NM - Fine body control system NAMC (Optimum Nutation & angular momentum control) for normal mode Roll/Yaw - Three-axis stabilized apogee boost maneuver with partially deployed solar generators

Table 3.1-2: Communication satellite AOCS features & performance (DASA-AOCS)

Low earth orbit Observation/Scientific Satellites		AOCS Development & Qualification	Launch dates
MOS-1 (Marine Observation Sat.)	1 engineering model 1 flight model	1983	FM1: 1987 FM2: 1990
ROSAT (Röntgen-Satellite)	1 engineering model 1 flight model	1984 - 1986 1988	01.6.90
EURECA (European Retrievable Carrier)	1 flight model	1988 - 1990	31.7.92
ASTRO-SPAS (Shuttle Pallet Satellite)	1 flight model	1988 - 1992	June 93, early 94

Table 3.1-3: Review of low earth orbit satellites (DASA-AOCS)

	Control Accuracy	Normal Mode Control Concepts
MOS-1 (Marine Observation Satellite)	Pointing Direction: 0.1° Stability: 0.001°/sec	Two bias momentum wheels in V-arrangement with magnetic unloading of accumulated angular momentum
ROSAT (Röntgen-Satellite)	Pointing mode: 3' Meas.Accuracy: 10" Scan Mode: 3'	Reaction wheel control with magnetic unloading of accumulated angular momentum
ASTRO-SPAS (Shuttle Pallet Satellite)	Pointing mode: 5" Scan Mode: 80"	Rigid body control with cold gas Use of GPS for Earth ref. frame determination
EURECA	0.9° each axis for normal mode	Cold gas control supported by magnetic control with disturbance torque compensation

Table 3.1-4: Earth/scientific satellite AOCS features and performance (DASA-AOCS)

Sensor	type	application-measurement range	Typical accuracy (1σ)	Weight [kg] *)	Power [W]	Manufacturer
Gyro	floated rate integrating (1 axis)	± 1°/sec	resolution: 1.12" random drift: 0.005°/h	1 + 1.5 = 2.5	4.5	FERRANTI
IR-Earth Sensor	conical scan (2 axes)	100 km to geo-synchr. altitude	LEO: 0.1° GEO: 0.05°	2.7	8	ITHACO
Sun-Sensor (SS)	4π Sterad. (2 axes)	2π	2° within ±20° of LOS	2x (0.2 + 0.2) = 0.8	2 • 1 = 2	TPD GALILEO
	Fine digital (2 axes)	± 64°	0.015°	9	8.6	DASA
Star-Sensor (STS)	CCD simultan. 3 star meas.	6° x 5° (magnitude 0 ÷ 7)	1" pointing 5" scan rate ≤ 5°/sec	(7 + 5) = 12 w.o. baffle	16	DASA
Magnetometer	flux gate (3 axes)	spherical (altitude: < 1000 km)	0.1 deg **) (earth dipole dir.)	0.5 + 1.5 = 2	0.7	ITHACO FOERSTER

*) including Electronics
**) ideal orbit and time reference

Table 3.2-1: Typical characteristics of attitude measurement equipment

Thrust Level [N]	Specific Impulse [sec]	Min.Imp.Bit [mNs]	Mass [kg]	Valve/Heater Power [W]
0.75 - 0.2	227 - 216	15 - 5	0.19	5/2.5
2 - 0.6	227 - 216	36 - 15	0.20	5/1
6 - 1.85	227 - 216	96 - 38	0.22	5/3.37
10 - 3	230 - 220	190 - 70	0.24	5/2.4
24 - 7.2	234 - 222	370 - 165	0.36	13/3
350 - 110	234 - 218	1100 - 350	1.80	30/--

Table 3.2-2: Characteristic parameters of (DASA) catalytic monopropellant hydrazine thrusters

Thrust Level [N]	Specific Impulse [sec]	Min.Imp.Bit [mNs]	Mass [kg]	Qualification Date
4	Fig. 3.2-2	13	< 0.25	92
10	Fig. 3.2-2	31	0.25	92
410	318 ± 2.5	---	2.7	93/94

Table 3.2-3: Characteristic parameters of DASA 2^{nd} generation liquid bipropellant thrusters

DIPOLE MOMENT [Am²] Linearity 1%	20%	SIZE [cm] length	diameter	WEIGHT [kg] Includes mounting blocks †	POWER * [W] Linearity 1%	20%
10	15	40	1.8	0.4 †	0.6	1.0
15	20	45	1.8	0.5 †	0.6	1.5
20	30	49	1.9	0.6 †	0.7	1.7
30	50	56	2.1	0.9 †	0.7	1.8
60	85	64	2.6	1.7 †	0.8	2.0
100	150	72	3.6	2.8	1.1	2.7
150	250	84	3.8	3.2	1.3	3.5
250	350	104	4.3	6.2	1.8	4.4
350	500	115	4.7	8.3	2.1	5.0
500	700	130	5.0	11.1	2.3	5.5
1,250	1,750	200	5.3	18.5	3.3	7.6
2,900	4,000	250	7.6	49.9	6.0	16.0

1 Ampere meter² = 1000 p−cm

*When a single winding is used, power doubles.

Table 3.2-4: Characteristic parameters of magnetic torquers (ITHACO)

Wheel diameter	cm	20	26	35	50*)
Angular momentum range	Nms	1.8...6.5	5.0...20	14...80	50...200
Max. reaction torque	Nm	0.2	0.2	0.2	0.2
Speed ***)	min^{-1}	6000	6000	6000	6000
Loss torque at max. speed**)	Nm	≤ 0.012	≤ 0.013	≤ 0.015	≤ 0.022
Power consumption:					
– steady state (depending on speed)	W	2...7	2...8	2...10	3...15
– max. power rating	W	≤ 60	≤ 80	≤ 100	≤ 150
Dimensions:					
– diameter A	mm	203	260	350	500
– height B	mm	75	85	120	150
Weight	kg	2.7...3.4	3.5...6.0	5.0...8.0	7.5...12
Environmental conditions: – operating temperature – vibration (sinusoidal) – vibration (random) – linear acceleration		suitable for satellites compatible with launchers such as ARIANE or Space Shuttle			

*) under development
**) with ironless motors
***) Max. speed of reaction wheels, nominal speed of momentum wheels (control range ± 10 %)

Table 3.2-5: Characteristic parameters of momentum and reaction wheels (TELDIX)

Thrusters	Thrust Level	Orientation	Location
Set A	4 N	tilted in the S/C (x,y)-plane --> zero z-component	on the corners of the cen tral cube
Set B	4 N	tilted in the S/C (x,y)-plane --> zero z-component	on the corners of the cen-tral cube
Set C	4 N	pointing in ± x-direction --> zero y- and z-component	on the edges of the S/C, parallel to the S/C y-axis

Table 4.2-1: Thrust levels, thruster locations and orientations

	thruster activation thruster no. 1 - 4	
torque requirement	1st half modulator time increment	2nd half modulator time increment
(-1, -1, -1)	(0,1,1,1)	(0,1,1,1)
(-1, -1, 0)	(0,0,0,1)	(0,1,1,1)
(-1, -1, 1)	(0,0,0,1)	(0,0,0,1)
(-1, 0, -1)	(0,0,1,0)	(0,1,1,1)
(-1, 0, 0)	(0,0,1,1)	(0,0,1,1)
(-1, 0, 1)	(0,0,0,1)	(1,0,1,1)
(-1, 1, -1)	(0,0,1,0)	(0,0,1,0)
(-1, 1, 0)	(0,0,1,0)	(1,0,1,1)
(-1, 1, 1)	(1,0,1,1)	(1,0,1,1)
(0, -1, -1)	(0,1,0,0)	(0,1,1,1)
(0, -1, 0)	(0,1,0,1)	(0,1,0,1)
(0, -1, 1)	(0,0,0,1)	(1,1,0,1)
(0, 0, -1)	(0,1,1,0)	(0,1,1,0)
(0, 0, 0)	(0,0,0,0)	(0,0,0,0)
(0, 0, 1)	(1,0,0,1)	(1,0,0,1)
(0, 1, -1)	(0,0,1,0)	(1,1,1,0)
(0, 1, 0)	(1,0,1,0)	(1,0,1,0)
(0, 1, 1)	(1,0,0,0)	(1,0,1,1)
(1, -1, -1)	(0,1,0,0)	(0,1,0,0)
(1, -1, 0)	(0,1,0,0)	(1,1,0,1)
(1, -1, 1)	(1,1,0,1)	(1,1,0,1)
(1, 0, -1)	(0,1,0,0)	(1,1,1,0)
(1, 0, 0)	(1,1,0,0)	(1,1,0,0)
(1, 0, 1)	(1,0,0,0)	(1,1,0,1)
(1, 1, -1)	(1,1,1,0)	(1,1,1,0)
(1, 1, 0)	(1,0,0,0)	(1,1,1,0)
(1, 1, 1)	(1,0,0,0)	(1,0,0,0)

Table 4.2-2: Correspondance between torque requirements and thruster activation

	S/C in orbit orientation 1		S/C in orbit orientation 2		
	Orbit normal	Orbit tangential	Orbit Normal	Orbit tangential	
Nominal	Set A or Set B	Set C	Set C	Set A	Set B
Backup	Set	Set A/B Set B/A	Set A/B or Set B/A or Set C	2 remaining ' thrusters of set A	2 remaining ' thrusters of set B

' residual pitch disturbance torque compensated by set C

Table 4.2-3: Thruster activation for orbit corrections

OB-CU	Kernel of the ICDS (On Board Computer Unit)	Ground Station I/F High priority telecommand decoding µ-Processor + memory Reconfiguration and safeguard memory	UPSE	Thruster Valve Drivers
Data Bus	Data Transmission Link	OBDH protocol for ML16/DS16 exchange On/Off command execution Analogic Thermistor	PSSE	Precision Sun Sensor Electronics
PFIU	Platform Interface Unit	Interface adaptation to AOCS sensors On/Off distribution for PF units Acquisition of PF units temperatures	HLC1	High Level Priority TC-commands
PFDU	Platform Distribution Unit	Power distribution and protection of PF Unit Pyro of the whole S/C I/F with PCU (Power Conditioning Unit) Battery protection HW PF heater (relays)	HLC2	High Level Priority Reconfiguration Commands
PLDU	Payload Distribution Unit	Power distribution and protection of P/L unit P/L unit temperature acquisition P/L heater relay On/Off distribution to P/L unit TM/TC of payload	SADA	Solar Array Drive Assembly

Table 4.3-1: Functional sharing and notation of ICDS H/W units

Equipment	Mass	Power	Supplier	Remarks
1 OBCU	12 kg	25 W	DASA	internally redundant
1 PFIU	6 kg	7 W	SEXTANT	internally redundant
1 UPSE	6 kg	6 W	AL	internally redundant
1 PSSA	9 kg	9 W	MBB	internally redundant
1 PFDU	13 kg	25 W	SEXTANT	internally redundant
1 IRS	2.7 kg	8 W	GALILEO	
1 RIGA	10 kg	18 W	FERRANTI	4 gyros, one scewed
2 FMW	18.4 kg	19 W	TELDIX	redundant set
2 RW	10 - 13 kg	8 W	TELDIX	redundant set
1 STS	12 kg	16 W	DASA	
Torque-rods	14.4 kg	12 W	FOKKER	3 coils, int. redundant, 350 Am^2

Table 4.5-1: Preliminary AOCS equipment mass and power budgets

Electric Propulsion for Lightsats: A Review of Applications and Advantages

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SUMMARY

The paper reviews the advantages and limitations of electric propulsion for lightsats characterized by a launch mass in the 300 to 800 Kg range. The considered systems include ion propulsion, arcjets, and stationary plasma thrusters for different applications such as drag compensation, orbit raising, fine trimming of orbital parameters, and various orbit transfers.

Electric propulsion provides substantial mass savings and turns out to be an enabling technology for certain lightsat missions. The constraints resulting from DC power limitations onboard small satellites are discussed, along with the implications on candidate technologies and system solutions. A review of near term prospectives of low power ion thrusters, for lightsats applications, concludes the paper.

1. INTRODUCTION

Recent advances in electric propulsion have finally led to consider its use for commercial geostationary communication satellites stationkeeping. For such function, ion thrusters, stationary plasma thrusters (SPT) and arcjets are currently being planned on board several new satellites. The main attraction of electric thrusters lies in their high exhaust velocity which is several times that of conventional chemical thrusters. The major benefit is represented by the propellant mass reduction for orbit control tasks.

Electric propulsion is also being considered for other near Earth propulsion applications, see e.g. [1], in particular for orbit transfer and manoeuvering of large spacecraft; and for scientific and interplanetary missions.

For what concerns the electric propulsion applications to lightsats, some initial work results from the open literature, see e.g. [2], but no specific plans are known to the authors about actually implementing electric propulsion onboard lightsats. This is quite surprising because, in view of the specific constraints of small satellites, electric propulsion may be considered an 'enabling' technology without which a number of applications would be almost unfeasible.

The paper reviews the key features of electric propulsion in the specific context of DC-power limited lightsats characterized by a launch mass range in the 300 to 800 Kg, which is most appropriate for a number of 'professional' applications, including civil and defense ones.

2. ELECTRIC PROPULSION FOR LIGHTSATS: MOTIVATIONS AND CONSTRAINTS

The main reason to consider electric propulsion is to reduce the total subsystem mass for propulsion-related tasks, thus maximizing the payload to launch mass ratio while staying within the launch mass bounds stated above. However, electric propulsion can be a viable approach only if the power requirements are also compatible with the DC power limitations, considering that the lightsat power plant is normally designed to support the payload operation in normal mode. We will limit our considerations to three E.P. technologies: ion propulsion, Stationary Plasma Thrusters, and arcjets.

In the low-power range of interest for lightsats, typical performance are given in Table 2.-1 which envelopes the characteristics of a number of existing devices. Two main Electric Propulsion applications are considered:

- orbit keeping, which includes drag compensation, fine trimming of orbit parameters and, for lightsats in synchronous orbit, station-keeping. These propulsion tasks are normally concurrent with payload operation. Therefore E.P. applications are mainly power limited, besides being also mass limited in order to result competitive with chemical propulsion. To exemplify, we will assume to allocate 30 Kg and 300 W to such E.P. functions;

- orbit manoeuvering, which includes orbit raising, circular to elliptical orbit transfer, and orbits circularization. These propulsion tasks are performed outside payload operation. Therefore the full lightsat DC power, normally used to supply the payload in normal mode, can be made available for Electric Propulsion. Typically, one may allocate 60 Kg and up to 1200 W of DC power to E.P. functions.

E.P. Technol.	Isp (sec.)	Spec.pwr (W/mN)	Spec.mass (Kg/mN)	Thrust (mN)
Ion	2600-3400	25-30	2-3	2-60
SPT	1400-1800	19-20	1.2	?-80
Arcjet	500-600	8-9	0.15	150-250

Table 2.-1 Typical characteristics of three E.P.technologies in the low-power range

Considering typical values from Table 2.-1, it can be seen that , for orbit-keeping tasks, arcjets are severely power limited and are further penalized by the poor Isp value. SPTs are good contenders to ion thrusters, in that they can provide higher thrusts for the same DC power at a lower mass, but the lower Isp value can be critical for long-duration missions. For orbit manoeuvering the available DC power is just at the limit where arcjets become interesting; but SPTs also appear to be good candidates, in view of the better Isp they can offer. Other considerations, such as the time to transfer, must then be taken into account, in the system trade-offs, to choose the right technology.

3. TYPICAL E.P. APPLICATIONS TO LIGHTSATS

3.1. Drag compensation for Observation satellites

Cost-effective observation missions can be conceived with individual satellites, launched on-demand, or based on small satellites constellations carrying SAR [3] or optical sensors. In both cases the mission requires very low orbit altitudes, say in the 280 to 600 Km range, to enhance the optical ground resolution or, in the SAR case, to reduce the transmitted power compatibly with the limited resources available on a small satellite.

At these altitudes the residual athmospheric drag becomes the limiting factor for the mission duration. Even assuming a slender platform design and an orientation of the appendages (i.e. solar arrays, antennas) such as to exhibit a minimum cross section in the direction of the flight path, the delta-velocity required to counteract the residual drag becomes rapidly unmanageable with chemical propulsion for mission durations greater than a few months: see Table 3.1-1. which is relevant to two typical lightsat configurations. The determining factor becomes the propellant mass, given the launch mass limits.

The high specific impulse of ion propulsion allows reducing, by a factor of about 10 w.r.t. chemical propulsion, the propellant mass required to impart a given total velocity increment. As a consequence, mission lifetimes of 5 years can be achieved, which is instrumental to implement small satellite constellations for 'permanent' observation missions. In comparison, the other E.P. technologies do not perform well in this application characterized by very high required velocity increments.

With ion propulsion the applied thrust levels are close to the drag force averaged during one orbit period, i.e. from 3 to 15 mN for a typical small SAR satellite intended to fly at orbit altitudes as low as 300 Km. Thrusts of this order can be modularly achieved by putting low power (< 10 mN) ion thrusters in parallel and implementing throttling or on-off modulation according to the required duty.

Orbit height (Km)	Drag f.ce avgd. over 1 orbit(mN) a)	b)	Delta-V 6 mo. (m/s) a)	b)	Delta-V 5 years (m/s) a)	b)
275	30	15	1155	385	11550	3852
300	21	10.5	815	272	8150	2721
350	14.2	7.1	551	184	5508	1840
400	11.7	5.9	457	152	4752	1522
450	8.9	4.4	345	115	3452	1152
500	7	3.5	270	90	2700	902

Assumptions:
* case a): S/C crossection 4 m^2, drymass 400 Kg;
* case b): S/C crossection 2 m^2, drymass 600 Kg;
* worst year sun activity;

Table 3.1-1 Drag forces vs. altitude and required delta-velocities for their compensation

Table 3.1-2 gives propellant mass and DC power requirements of ion propulsion vs. orbit altitude for a prospective small SAR satellite. For the considered thrust levels, the DC power requirements are compatible with spacecraft allocations. In conclusion, ion propulsion has to be considered an enabling technology to implement long duration observation missions in very low Earth orbits.

Nevertheless the operating time, of the order of 40000 hours, is of concern and should be properly addressed by further developments. Efforts to further improve the efficiency of low power thrusters are also strongly reccomended.

Orbit height (Km)	Propell. mass (Kg)(1)	Thrusters N°/ avge orb.duty	Avge DC power (W),(2)
275	90	2*8mN/0.94	450
300	62	2*8mN/0.66	320
350	41	8mN/0.88	240
400	34	8mN/0.74	200
450	25	8mN/0.55	150
500	20	8mN/0.44	120

Note 1): based on 2800 s. Isp
Note 2): based on 30 W/mN

Table 3.1-2 Drag compensation ion propulsion requirements for system b) of Table 3.1-1

3.2 Fine trimming of constellations orbital parameters

An increasing number of small satellites constellations is being proposed [4] for advanced communication and remote sensing applications: most constellations have multiple satellites in one or multiple orbital planes; low altitude orbits, typically 400 to 1500 Km, are normally envisaged, with a few cases considering altitudes as high as 10000 Km.

Managing these complex systems will require copying with the launch systems orbit injection dispersions, as well as maintaining the satellites relative phasing during the orbital lifetime against external disturbances. The quantization of these effects, in terms of propellant mass, depends strongly from the launch vehicle characteristics, orbit type and mission duration, and will not be attempted here. The push towards launch cost reduction might probably imply, in the near future, the use of less accurate L.V. and greater injection dispersions. There is also a clear trend to require longer orbital lifetimes.

Both factors will tend to increase the propellant mass allocated for orbit control: this will impact negatively the system economics, also in view of the multiplying effect due to the large number of satellites in these constellations. Electric propulsion may be considered in alternative to chemical propulsion for the potential mass saving it can provide, specially in presence of high orbit injection dispersions and long orbital lifetimes. Constellation phasing is not a time-limited propulsion task, which will enable using low thrust, high specific impulse, E.P. systems. Suitable strategies for constellation phasing maintenance can be conceived, possibly aided by autonomous navigation systems, to fully exploit the E.P. characteristics.

3.3 Stationkeeping of small geosynchronous satellites

Small satellites in geostationary orbit can perform usefull communication, meteorological and observation missions. Clusters of two to four spacecraft can form a distributed satellite system having enhanced performance w.r.t. a single satellite of an identical total mass.

Assuming the availability of a launch vehicle capable of directly injecting the satellites in synchronous orbit (such as, e.g. the Russian Proton rocket) small spacecraft relying exclusively on E.P. can be conceived.

The absence of a bulky apogee motor greatly expands the spacecraft design freedom, including the positioning and orientation, or canting angle, of the E.P. thrusters. Spacecraft incorporating several small thrusters may implement both orbit control and external torques compensation for attitude control.

Table 3.3.-1 gives the essentials of an initial concept for a high performance small geostationary satellite. Basic features are the low power/low thrust ion-thrusters, the resulting overall low E.P. system mass, the high achievable payload to satellite launch mass ratio, the absence of any kind of ,chemical propulsion.

* Satellite launch mass (Kg)	:	600
* Payload mass (Kg)	:	240
* Payload/launch mass ratio	:	> 0.4
* Orbital lifetime, years	:	15
* Rqrd. total velocity increment(m/s) N-S/S.K.	:	700
E-W/S.K.	:	70
* Ion thrusters number	:	10 to 12
* Thrust levels(mN) N-S/S.K.	:	8mN (2*4mN)
E-W/S.K.	:	4mN (2*2mN)
* Total prop.mass (Kg)	:	< 18
* Avge dayly thrusting time N-S/S.K.	:	2.66 hours
E-W/S.K.	:	32 min.
* E.P. system mass (Kg)	:	< 30 (est.)
* E.P. peak DC power (W)	:	240
* Dayly energy needs(KWh)	:	< 0.7
* Feasible control torques in pitch,roll,yaw	:	<= 8*10-3 Nm

Table 3.3-1 Main features of a small geostationary satellite using small ion thrusters

3.4 Circular orbit raising

Several small launch systems are becoming available, capable of direct injection in L.E.O. of spacecraft with launch masses in the 250 Kg to one ton range. The L.V. payload carrying capability decreases rapidly, however, with circular orbit altitude injection: therefore it may be preferable to inject in a relatively low orbit a more massive payload, comprising a satellite and a boost motor, using the latter for orbit raising. Typical cases may be represented by a coplanar orbit raising from a 400 Km initial orbit to a final circular orbit of 800 to 2000 Km: the resulting velocity increments are given in Table 3.4-1.

Chemical boost motors are normally envisaged for cost and technology maturity reasons, but their mass constitutes a significant fraction of the L.V. payload and will reduce, in proportion, the satellite mass and its payload.

Electric propulsion can be considered in alternative, also because during orbit raising the payload is not operative: the full satellite DC power is, thus, in principle available for E.P., with the constraints given by earth eclipses.

Continuous thrusting is therefore assumed only during 50 % of the orbit period, with the thrust vector tangent to the orbit plane. Table 3.4-1 compares approximately estimated values for propellant mass and orbit tranfer times for the candidate E.P. technologies w.r.t. a chemical one.

	Final orbit altitude (Km) (1) 800	1400	2000
* Delta-V.(m/s)	217	510	772
* Prop.mass (Kg),(2):			
- chemical (4)	48	120	190
- ion (5)	4.6	11	17
- SPT (6)	8.2	19.5	29.6
- Arcjet (7)	26.5	64.6	100
* Trip time (days) (3):			
- chemical	<1	< 1	<1
- ion	76	178	270
- SPT	51	120	180
- Arcjet	21	50	75

1) 400 Km initial circular orbit;
2) 600 Kg spacecraft
3) thrust levels: ion= 40 mN;
SPT= 60 mN;
arcjets= 150 mN;
4) Isp=280 s; 5) Isp=2800 s;
6) Isp= 1600 s; 7) Isp=500 s;

Table 3.4-1 Electric Propulsion systems for Orbit Raising

Table 3.4-1 shows that low power arcjets are good candidates for high altitude differentials, providing good mass savings with reasonable transfer times. On the contrary, ion thrusters and SPTs are marginal and can be only considered for small altitude increments, otherwise too long tranfer times will result. In conclusion, low thrust electric propulsion can be considered for circular orbit raising of small satellites when a short transfer time is not mandatory.

When orbit raising must be implemented in a very short time, as for example with satellites launched on-demand, then chemical propulsion is still the preferred approach.

3.5 Circular to elliptical orbit transfers

Elliptical inclined orbits are being considered with increasing interest for numerous applications, and there are plans to realize small satellite constellations exploiting the particular features of such orbits. However, the injection capability of small launchers in elliptical, high energy, orbits is normally rather poor, and one has , again, to rely on boost motors to implement the final orbit transfer, for which two important considerations apply.

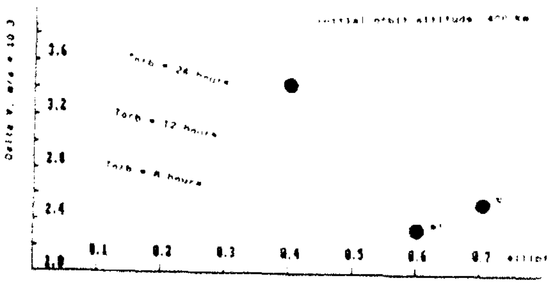

Fig 3.5-1 Circular to elliptical orbits tranfers
Delta V vs. ellipticity and orbital period

First, very high delta-velocities are required, see Fig. 3.5-1. Second, commercial communication satellites injected in elliptical orbits can be planned and launched well in advance of the need date. Long trip times are, thus, relatively unimportant but mission complexity, costs, reliability and operational considerations may put an upper bound to the maximum acceptable trip time.

Electric Propulsion alternatives to chemical are SPTs and arcjets, with the same characteristics given in Table 3.4-1. Simplified computations for transfers of a 500 Kg spacecraft in transfer orbit from an initial 400 Km circular orbit to three destination elliptical orbits of 8, 12 and 24 hours periods were performed.

Near-ideal transfer was simulated by firing thrusters for 1/4 of the orbit period about the perigees until the final apogee was reached, then firing the thrusters for 1/4 of the orbit period until the final perigee was also reached. Propellant mass required by SPTs and arcjets are reported in Table 3.5-1.

	Destination Orbit (1) a)	b)	c)
** Ideal Tranfer:			
- Delta-V. (m/s):	2350	2550	3450
- T.O.mass/ propell.mass (Kg)(2):			
chemical	1157/657	1243/743	1714/1214
arcjet	800/300	832/332	996/496
SPT	579/79	586/86	620/120
** Non-ideal Transfer, Arcjet:			
- T.O. mass/ propell.mass (Kg)(3):	858/358	898/398	1105/605
- Trip time (days)	363	394	530

Characteristics of Destination Orbits:
a) Apogee:32430 Km; Perigee:8107 Km; Period: 8 hours;
b) Apogee:45150 Km; Perigee:7968 Km; Period:12 hours;
c) Apogee:59030 Km; Perigee: 25298 Km; Period: 24 hours;
Note 1) Starting orbit: circular, 400 Km;
" 2) Satellite drymass: 500 Kg
" 3) Including Delta-V increment (15%) for non ideal transfer

Table 3.5-1 E.P. alternatives to chemical propulsion for circular to elliptical orbits tranfers

For the arcjet case, a non-ideal transfer was also simulated, and the propellant mass recomputed taking into account a 15 % worsening in the delta-velocity required due to the non-ideal tranfer conditions. As can be seen the trip times are quite high in all cases: a 30 % reduction can be achieved increasing the arcjet thrust to 220 mN, which would nevertheless imply the availability of 1.8 Kw DC power. This might necessitate an auxiliary array in the E.P. boost section, impacting costs.

There are several additional positive and negative factors, to be taken into account, which may affect the trip time; but in conclusion the feasibility of using low power arcjets in power limited lightsats for low altitude circular to elliptical orbits coplanar transfers has to be considered still problematic.

3.6 Orbits circularization

This propulsion task refers to satellites, injected in elliptical transfer orbit by a L.V., whose final destination orbit is circular with a radius normally coincident with the T.O. apogee. Typical cases are geostationary orbits and medium altitude circular orbits of about 10000 Km as envisaged, for example, by TRW's Odyssey and ESA's MAGSS-14 constellation [5].

Electric propulsion is compared to chemical in Table 3.6-1. for these two typical missions. Arcjets are found to perform acceptably well in this case, since trip times around 200 days can be achieved along with significant mass savings w.r.t. a chemical propulsion alternative. The better mass saving achievable with SPTs is, instead, paid with an excessive trip time duration.

** T.O. characteristics:		
- Perigee (Km)	6678	6778
- Apogee (Km)	42164	16728
** Destination Orbit characteristics:		
- Radius (Km)	42164	16728
- Velocity increment, ideal transf.(m/s):	1476	1174
** T.O.mass/propell.mass (Kg) (1)(2):		
- chemical	844/344	760/260
- arcjet	700/200	655/155
- SPT	555/55	544/44
** Trip times (days):		
- chemical	<1	<1
- arcjet	227	182
- SPT	n.a.	453

Note 1): Satellite drymass 500 Kg;
" 2): Including a 1% increase in delta velocity due to non-ideal tranfers

Table 3.6-1 E.P. alternatives for Orbit Circularization

4. REVIEW OF SYSTEM NEEDS

From this overview two E.P. technologies appear to have a future on power limited lightsats:

- small ion thrusters for drag compensation, limited orbit raising, orbit circularization and fine trimming of orbital parameters of small satellites. The required thrust levels are in the 2 to 10 mN range. a modular approach enabling parallel operation of thrusters is also required. Full thrust control, over a 30% to 120 % range of the design thrust level, is an important design requirement. An overall specific power of 40 W/mN maximum (30 W/mN as a goal), and a specific system mass of less than 1 Kg/mN maximum (0.7 Kg/mN as a goal), including power supply and logic should be set as near term development objectives. Isp values better than 3000 sec., and very long lifetimes, of the order of 40000 hours, are also primary requirements;

- low power arcjets in the 1 to 1.2 Kw range, for orbit manoeuvering tasks. For such devices efforts should be devoted to possibly bring the Isp closer to 600 sec. for thrusts in the 150 to 250 mN range, with an overall specific power better than 8 W/mN including power supply.

On the other hand stationary plasma thrusters do not seem to have a clear role for lightsats, at least in their present power and thrust level range. Scaling down SPTs to low thrust values (say below 10 mN), while keeping unchanged their excellent performance, in particular efficiency and simplicity, is yet unproven but might be worth being investigated.

5. NEAR TERM TECHNOLOGY PROSPECTIVES OF LOW POWER ION THRUSTERS

For the considered thrust range, new devices based on the Electron Cyclotron Resonance (ECR) [2] are presently under evaluation, aiming to further improve the ion thruster's performance.

The ECR technique allows operating over a wider thrust range by changing the mass flow rate in the discharge chamber enhancing the electrical efficiency and gas consumption over a wider pressure range than the conventional RF and Kaufman technologies. ECR appears particularly indicated for small size ion thrusters, for which it is also easier to achieve the optimum static magnetic field necessary for resonance.

The ECR technique consists in applying a static magnetic field , orthogonal to the direction of an oscillating RF field, so that electrons are forced to rotate within the thruster's discharge chamber, around the magnetic field lines at a cyclotron frequency given by: $Fc = eB/2\pi m$. The mean energy per collision, transferred to an electron, is:

$$Wc = (e^*E^2/4m)(1/(4\pi^2(F-Fc)+(1/\tau^2))),$$ with:

B= magnetic field; e= electron charge; m= electron mass; E= applied electric field peak amplitude; τ= mean time between two consecutive collisions, inversely proportional to gas pressure; F= frequency of the applied RF field.

At resonance F=Fc, and the quantity Wc assumes its maximum value: $Wc = (eE\tau)^2/4m$, which depends only on collision interval and electric field peak value. Clearly the maximum benefits of the ECR effect is achieved in the low pressure range, being the collision frequency proportional to the operating gas pressure.

For a ion propulsion system in the millinewton range a RF excitation in the VHF range proves advantageous for the following considerations:

- it is possible to use a low level static magnetic field to reach the resonance condition, e.g. 36 Gauss for resonance at 100 MHz. A lower thruster mass can be thus achieved;

- since the wavelenght is much higher than the chamber dimensions, problems arising from energy propagation into plasma can be avoided. In case of VHF excitation, plasma generation is infact achieved through a "near field" mechanism so that no cut-off frequency exists;

- it is possible to use coupling electrodes, for the VHF RF field, outside the discharge chamber, due to the limited power losses for irradiation and absorption in the chamber walls, compared to the usefull power transferred to the plasma.

Considering a thrust level in the 2 to 10 mN range, and a cylindrical discharge chamber of 3 to 5 cm radius, three conditions are required to fully exploit the ECR benefits:

1) $F >> 1/\tau$, i.e about 2 MHz considering a pressure of 10^{-4} Torr and an electron energy of 13eV. This condition ensures that electrons turns around the magnetic field many times between consecutive collisions;

2) $R < 1$ cm, where R is the gyration radius of the electron in presence of a magnetic field directed along the chamber axis. This condition ensures that the electron trajectories are contained in the discharge chamber;

3) EM field wavelenght >> 3÷5 cm. This condition avoids propagation problems in the plasma.

In summary, the ECR technique is expected to provide substantial improvements in low power ion thrusters performance, enhancing the advantages of traditional RF techniqes in combination with others typical of the Kaufman technology. The main features of the ECR technique are summarized as follows:

- ECR sources do not have components subjected to wearout, as cathodes or accelerating electrodes, in the plasma chamber.

Problems of sputtering erosion inside the chamber are thus eliminated, impacting positively the thruster lifetime. Moreover, it is possible to realize the discharge vessel with materials having high secondary electron emission coefficients, scarcely sensitive to erosion and capable of reducing electron losses from plasma towards the vessel walls.

- The plasma produced by the ECR is highly uniform, both in density and temperature. Achievable electron densities are of the order of 10^{12} electrons/cm^3 at pressures in the range of $10^{-4} \div 10^{-5}$ Torr, thus facilitating beam extraction optimization.

- The ECR technique is specially advantageous for ion thrusters operating at low thrust levels (2÷10 mN) where it reduces considerably the instability phenomena inside the plasma. Low thrusts are related to small chamber dimensions, which simplifies the realization of a uniform magnetic field in the discharge vessel, and requires less DC power. A small chamber also requires small electrodes for electromagnetic energy transfer, with lower losses due to EM irradiation and vessel walls absorption.

Furthermore, for constant average beam current densities, thruster throttling is accomplished keeping the mass flow rate to beam area ratio constant. A reduced flow rate is associated with a pressure regime within the discharge chamber where ECR proves more efficient and stable than other traditional excitation techniques.

- The grid extraction system can be designed and optimized on the basis of the enhanced plasma characteristics typical of the ECR technique. In this direction PROEL is investigating also new technologies (patent pending) for the realization of grids with a geometry having a high transparency factor to ions, while maintaining good mechanical resistance and the capability to survive to sputtering erosion.

The use of a high melting point refractory material, coated with suitable antisputtering protection layer, is the solution to long lifetime requirements.

6. CONCLUSIONS

Electric propulsion, and in particular advanced ion propulsion, is an enabling technology for the feasibility of very low Earth orbit missions, in power-limited lightsats, where drag is the dominant effect.

Furthermore, low power ion thrusters are a good system alternative for propulsion tasks requiring the steady application of low thrust levels over extended periods of time to implement a given velocity correction. Nevertheless very long lifetimes, of the order of 20000 to 40000 hours, must be reliably demonstrated. Improvements in mass and power efficiency of such low power thusters would also benefit the lightsats design.

Low power arcjets, in the 1 Kw range, have also promising applications for certain orbit manoeuvering tasks, and their development should be further pursued.

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An Overview of DARPA's Advanced Space Technology Program

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1. INTRODUCTION

In the three decades that have elapsed since the beginning of the space age, satellites have increased in performance and capability by orders of magnitude. Demands for more capacity have, in many cases, been met by merely increasing the size and weight of the satellites, with more channels for communication or more memory or more processing capability, but still representative of the existing state-of-the-art. In general, the size of satellites has grown significantly over the early trail-blazer versions with their admittedly limited capabilities. Yet, there are indications that we still have a long way to go before the ultimate limits are reached in "making things small."

In the present economic climate, it appears certain that there will be significant cutbacks in most areas of the defense budget, including military space systems. The Department of Defense (DoD) and the defense industry will be expected to design and field more advanced and more efficient space systems. These newer systems must, and will, make even more use of miniaturized, high technology components than are incorporated in our current stable of satellites. Using this philosophy, future systems can be expected to have equal, if not greater, capabilities than most of the current systems on a unit weight basis. Upon implementation of this philosophy of reduction in the size and cost of both satellites and their booster rockets, we will have made major steps toward synergistically satisfying emerging national needs for responsive and reconstitutable space systems.

The Defense Advanced Research Projects Agency (DARPA) is the central research and development organization of the DoD and, as such, has the primary responsibility for the maintenance of U.S. technological superiority over potential adversaries. DARPA's programs focus on technology development and proof-of-concept demonstrations of both evolutionary and revolutionary approaches for improved strategic, conventional, rapid-deployment and sea power forces and on the scientific investigation into advanced basic technologies of the future. DARPA can move quickly to exploit new ideas and concepts by working directly with industry and universities.

For four years, DARPA's Advanced Space Technology Program (ASTP) has addressed various ways to improve the performance of small satellites and launch vehicles. The advanced technologies that are being and will be developed by DARPA for small satellites can be used just as easily on large satellites. The primary objective of the ASTP is to enhance support to operational commanders by developing and applying advanced technologies that will provide cost-effective, timely, flexible and responsive space systems. Fundamental to the ASTP effort is finding new ways to do business with the goal of quickly inserting new technologies into DoD space systems while reducing cost. In our view, these methods are prime examples of what may be termed "technology leveraging."

The ASTP is a multidisciplinary technology development effort aimed at quickly exploiting advanced technologies. Included in the program are:

- Sponsorship of the initial launches of the Pegasus Air-Launched Vehicle (ALV) and the development of a Standard Small Launch Vehicle (SSLV);
- Development of enabling technologies subdivided into satellite support subsystems, satellite payloads and communication terminals;
- Development and launch of small satellites for demonstration to and evaluation by the military Services. Included in this area are the Multiple Access Communications Satellite (MACSAT) program and the Microsat program;
- Development of a standardized, multimission common bus under the Advanced Technology Standard Satellite Bus (ATSSB) program; and
- Development of two advanced technology satellite demonstrations scheduled for launch in the mid-1990s. The Advanced Satellite Technologies for EHF Communication (ASTEC) program will develop and launch two EHF payloads. The Collaboration on Advanced Multispectral Earth Observation (CAMEO) program will develop and launch a multiple-use, remote sensing, multispectral payload.

The ASTP has initiated over 50 technology projects, many of which have been completed and transitioned to users. The objectives are to quickly qualify these higher risk technologies for use on future programs and reduce the risk of inserting these technologies into major systems, and to provide the miniaturized systems that would enable smaller satellites to have significant -- rather than limited -- capability. Only a few of the advanced technologies can be described here, the majority of which are applicable to both large and small satellites.

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2. ENABLING TECHNOLOGY PROGRAMS

2.1 Subsystems

A significant portion of the ASTP effort is devoted to the development of satellite subsystem technologies. Subsystems such as power supplies, attitude determination and control devices, communications, computer processing and memory storage, propulsion units for stationkeeping and repositioning are, of necessity, present on virtually every satellite (whether "light" or "heavy"). All of these items contribute to the weight of the satellite on-orbit; thus, major reductions in weight and cost not only benefit the satellite, but also aid in reducing the size and cost of the booster vehicles that put them in orbit. The ultimate objective is to minimize the size and maximize the efficiency of launches vehicles and the satellite bus so that the "business end" of the the space system -- the payload -- comprises a much more significant fraction of the overall satellite weight.

2.1.1 Lightweight Reaction Wheel

The Lightweight Reaction Wheel (LRW) is a magnetically suspended reaction wheel with redundant electronics that will provide five times the momentum of existing units at the same weight, with growth potential to ten times the momentum, by using magnetic bearings in lieu of ball bearings and faster rotational speed. The LRW requires 25 percent less power and can be used on all satellites; its higher speed and reduced bearing vibration potentially can benefit operation of onboard sensors (e.g., less "blurring" of a sensor's image). This effort is aimed at the design and delivery of a prototype LRW for testing and subsequent use on a satellite. The prototype LRW will be flown on an upcoming Air Force Space Test Program flight.

2.1.2 Miniature Global Positioning System Receiver

This effort is a technology development project to design, fabricate, develop, deliver and demonstrate, in an operational environment, a space-qualified, multichannel Global Positioning System (GPS) receiver than can be used to support autonomous navigation onboard spacecraft (Figure 1). The GPS receiver will enable the simultaneous reception of multiple GPS navigation signals, thereby providing greater position accuracy in comparison with currently available receivers that sequentially receive the multiple GPS navigation signals. The use of gallium arsenide (GaAs) technology has enabled this more "powerful" receiver to fit in a much more compact package (one-tenth the size of previously space-qualified receivers). The GPS receiver is scheduled to "fly" on several space missions beginning in 1993.

2.1.3 Attitude Determination, Control and Navigation System

A single, fully integrated guidance, navigation and control system is being developed under the Attitude Determination, Control and Navigation System (ADCNS) project. It will use low cost star trackers, fiber optic gyros, generic VHSIC space-borne computers and the ASTP-developed GPS receiver (see paragraph 2.1.2). The strap-down star tracker is designed to determine attitude by recognizing star patterns via an acquisition search pattern (a slow attitude change maneuver) and maintain that attitude determination in any orientation via nearly continuous star tracking. Low cost, highly reliable fiber optic gyros provide attitude determination during high slew rate maneuvers, smoothing between star sightings and rate stabilization. Spacecraft autonomy is maintained by using the GPS for navigational updates. Advantages of this project include a star pattern recognition algorithm that eliminates the need for an initial attitude acquisition sensor (i.e., no sun or earth sensor is required) and generic applicability to all three-axis, stabilized spacecraft.

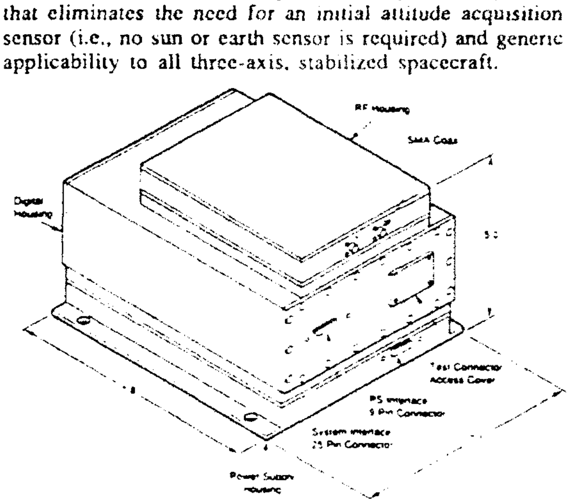

Figure 1. Miniature GPS Receiver

2.1.4 Inflatable Torus Solar Array Technology

Solar arrays typically provide spacecraft power. At present, as the amount of power required increases, the size and weight of the solar arrays increase -- a situation that necessarily constrains the capabilities of payloads on small satellites. The ASTP's Inflatable Torus Solar Array Technology (ITSAT) project aims to make possible space missions that are otherwise impossible by providing increased available power while maintaining small launch volume and weight. An inflatable, self-rigidizing torus structure supports the solar cell blanket. The modular design allows for easy insertion of new solar cell technology including thin film arrays. The current ITSAT under development has design goals of 100 watts/kg and 120 watts/m^2.

2.1.5 Miniaturized, Low-Power Parallel Processor

The basic goal of the Miniaturized, Low-Power Parallel Processor technology development project is the reduction, by an order of magnitude, of onboard processor size, weight and power consumption for space-based sensor systems. Such processors must employ massively parallel architectures with large numbers of processing elements in order to achieve the high throughputs required (up to tens of billions of operations per second or more). The approach for achieving this goal is to use three-dimensional, hybrid wafer scale interconnect and packaging technology. In this concept, individual modules with multiple, unpackaged semiconductor chips are compactly interconnected into a high density package with substantial weight and power savings over existing packaging approaches (Figure 2).

2.1.6 Magnetic Disk Mass Memory

Rotating disk memory subsystems have been a leading technology product for many years. Although not widely applied to space applications, they represent state-of-the-art for high density, mechanical memory subsystems. They are less complex and have fewer potential mechanical failure points than traditional spaceborne tape recorder systems. The objective of the Magnetic Disk Mass Memory project is to develop a magnetic rotating disk memory subsystem that will provide up to 1 Gigabyte of high-speed data storage. The effort will review available optical and magnetic disk equipment, determine the

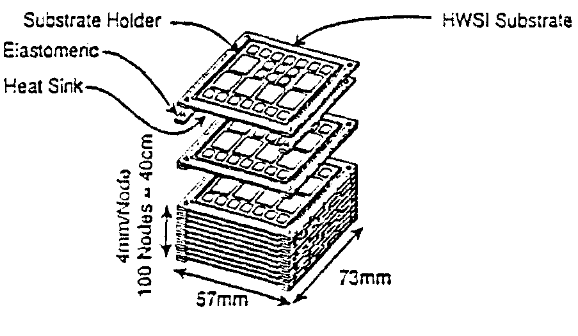

Figure 2. Miniaturized, Low-Power Parallel Processor Using 3-D HWSI

physical and environmental requirements for operation aboard a spacecraft, and design and fabricate an enclosure suitable for space flight.

2.2 Payload Technologies

In addition to providing more room for payloads on satellites, we intend to increase the number of bits or pixels per pound for communications and optical payloads. Several projects have been initiated to do just that. In this section, we discuss some of our EHF, UHF and laser communication initiatives and several of our optical technologies initiatives.

2.2.1 EHF

2.2.1.1 Payload Concept

The most promising, new, near-term technology for protected communications employs the EHF band (approximately 44 GHz uplinks and approximately 20 GHz downlinks). The ASTP has several technology projects underway that synergistically complement and support other DoD work in the EHF arena, including the application of advanced technologies that would significantly reduce the size, weight, volume and power requirements of existing EHF systems. We are striving for a 65 percent reduction in required power compared with the present technology base. Within five years, further technology advances should provide an additional 30 percent reduction in weight and power. Presently, our studies indicate a payload weight of approximately 67 pounds for a 35-channel, low data rate (LDR) EHF package is possible (compared to the existing 225-pound package). A highly efficient, 2 watts, 20 GHz transmitting, high-power amplifier yields an overall efficiency of at least 35 percent; it uses a recently developed (DARPA-sponsored) GaAs permeable base transistor. Also being developed for this project are a high-speed signal processor, lightweight signal generator and lightweight scanning antenna. The 44 GHz scanning antenna has a variable beamwidth, making it capable of operating in elliptic, as well as circular, orbits.

2.2.1.2 Spherical Lens Antenna

The Spherical Lens Antenna project has as its objective the development of a wide field-of-view (WFOV), electronic scar ing, multibeam EHF antenna capable of nulling interfering signals and operating in a variety of orbits, including those that are highly elliptical. The antenna's radiating aperture is a sphere of solid dielectric that provides a graduated index of refraction to form the radiating aperture. Feedhorns are arranged with equal spacing on a concave spherical surface adjacent to the dielectric sphere. To form an electronically scanned antenna, any one of a large number of feeds is excited through an interleaved, switched tree network, which can combine clusters of horns to perform complex nulling or can be used for simple switched beam operations. The size of the lens depends on the minimum gain requirement (e.g., for 25 dB minimum gain, the lens would measure approximately 2.5 inches in diameter at a frequency of 44.5 GHz). The estimated size of the antenna (lens, horns and switch tree) is 12-inch diameter cylinder and 10 inches long.

2.2.2 UHF

The most widely used and, unfortunately, most vulnerable military satellite communication (MILSATCOM) frequency band is UHF. The Multiple Path, Beyond Line-of-Sight (MUBL) Communication effort is aimed at providing interference-resistant voice communications among affordable, handheld UHF terminals having approximately 5 watts of RF output. The concept provides a single-hop capability (as seen by the user terminal); it may be thought of as an amplifying ionosphere. No satellite crosslinking is contemplated. When more than one satellite is in view of the communicating terminal pair, there are multiple propagation paths. The modulation and coding system is designed to support this and resist interference from other satellites. In addition to satellites, MUBL can be mounted on high altitude balloons, mountaintops or unmanned aerial vehicles. Successful implementation of these concepts could substantially alleviate shortcomings noted in the recent Desert Storm operations.

2.2.3 Satellite/Submarine Laser Communications Initiatives

A number of laser communications studies and demonstrations are underway or have been completed by the ASTP. Their common objective is to develop reliable, two-way laser communications systems. For submarines having expanded operational depth and speed envelopes, laser communications provides the potential for timely message delivery at useful transmission rates. Current submarine communications can be enhanced through the use of blue-green laser technology. Submarine laser communications systems will require small, lightweight, prime power efficient lasers and narrow spectral band, wide field-of-view optical filters. The ASTP has several blue-green laser projects that address these challenges. At present, these development efforts focus on the Cesium Atomic Resonance Filter (Cs ARF) and alternate narrow bandpass filters, as well as several laser transmitter technologies. All of these technologies are being developed for rapid insertion into candidate spaceborne laser communications systems.

2.2.3.1 Cesium Atomic Resonance Filter

The Cs ARF project is designed to investigate enabling technologies to support satellite-to-submarine laser communications. Specific areas of investigation are satellite system concept definition and uplink receiver breadboard design development and testing. The Cs ARF receiver approach utilizes a unique side-coupled Cs ARF as the basis for the 4-cell array (Figure 3). This approach provides very good broadband signal rejection with exceptionally high inband signal throughout. Our findings to-date have resulted in an order of magnitude improvement in cell efficiency from 4 percent to over 47 percent.

2.2.4 Optical Initiatives
Several research contracts for optical studies and breadboard demonstrations are completed or ongoing. The objective of these projects is to advance the state-of-the-art in satellite technologies for electro-optical sensor systems.

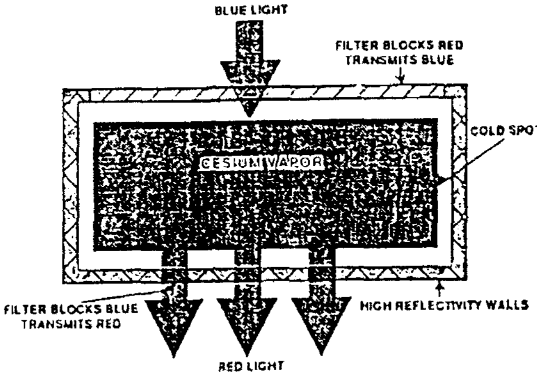

Figure 3. Cs ARF (Cross-Section)

2.2.4.1 Phased Array Mirror, Extendable Large Aperture
Aperture size of present space-based optical systems is limited by cost and availability of a vehicle to launch a large, massive, primary mirror. The Phased Array Mirror, Extendable Large Aperture (PAMELA) project is developing the technology to build a mirror composed of lightweight segments that can be folded for launch and automatically space-deployed to form a large aperture. (See Figure 4). The technology will develop and integrate segment sensors and position actuators with control algorithms for accurate remote deployment and active surface control. Miniature sensors on all adjacent edges sense relative segment offset. These sensors, considered key to the technology, have been demonstrated.

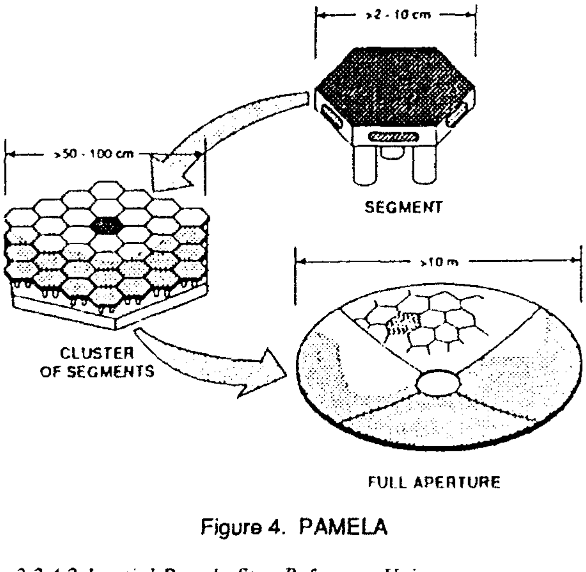

Figure 4. PAMELA

2.2.4.2 Inertial Pseudo-Star Reference Unit
The Inertial Pseudo-Star Reference Unit (IPSRU) project is developing a single unit for precision pointing and stabilization of optical payloads. It will compensate for mechanical flexural movements, thereby allowing the incorporation of less stiff and lighter structures. The IPSRU synthesizes the equivalent of an inertial star (an optical probe beam) that is injected into the path of a payload's telescope (Figure 5). This inertial star provides the means for closed-loop payload focal plane stabilization, which will eliminate the effects of jitter and transient distortions on the focal plane and the telescope structure.

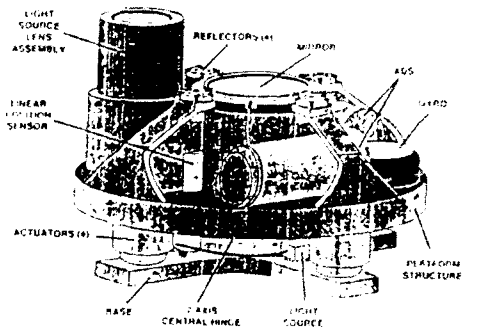

Figure 5. IPSRU

2.2.4.3 Electron Tunneling Sensor
Sensors for a wide range of signals are of primary importance for advanced systems in aerospace guidance and control, robotics, target imaging and other important applications. The ultrasensitive electron tunnel sensor, recently developed at Jet Propulsion Laboratory, has applications in advanced accelerometers, hydrophones, magnetometers and room-temperature infrared detectors. The tunnel sensor is based on the quantum-mechanical electron tunneling mechanism used in the scanning tunneling microscope, which won the 1986 Nobel Prize for Physics. A compact tunnel sensor with the size and mass of a penny has been micromachined from a silicon wafer. Because the tunnel sensor can be fabricated in silicon, it is possible to fully integrate the tunnel sensor with its microelectronics in a monolithic silicon package.

2.3 MILSATCOM Terminal Technology Development
DARPA's IMPACT (Insertion into MILSATCOM Products of Advanced Communications Technologies) Program is a multidisciplinary technology development effort aimed at phased insertion of advanced technologies into MILSATCOM terminal systems. The fundamental goal of this program is to reduce the life cycle cost of the MILSATCOM terminal segment with associated reductions in terminal size, weight and power consumption and enhancements in performance, reliability and capabilities. The program addresses broad technology efforts that span all MILSATCOM terminal programs with technology insertion initiatives, retrofits and upgrades, as well as enabling technology developments in support of next-generation terminals.

MILSATCOM systems currently encompass a diverse multitude of terminals that are designed to interface with several different types of satellites operating in three MILSATCOM bands. Each frequency band offers particular advantages and complementary characteristics to the overall defense space architecture.

In addition to this diversity, the terminal population incorporates numerous distinct system implementations, including manpack and man-portable terminals; mobile

Significant technological advances occurred during the time that most of the infrastructure was acquired; however, it is not cost effective for DoD to procure a *new* suite of terminal systems despite the enhanced performance, novel capabilities and other advantages that new technologies have to offer. Instead, innovative product and process strategies are needed to modernize and upgrade the existing infrastructure by inserting advanced technologies into current systems. In this way, new technologies *can* be exploited in a manner that leverages our prior investments in the MILSATCOM architecture.

Since a fundamental goal of DARPA's IMPACT Program is successful and immediate technology transfer, close coordination has been established with key organizations and programs in the MILSATCOM community. These key players include system users, technology developers in Service laboratories, terminal product management offices, the executive management for the MILSATCOM Architecture, the Defense Science and Technology Program, and agents for execution of the program.

IMPACT will work closely with the MILSATCOM terminal community and with technology developers. By conducting an integrated, coherent effort, IMPACT will synergistically leverage ongoing work within this community and within industry to achieve the product- and process-oriented goals of the program.

IMPACT will serve as a testbed and role model program for the new era of defense acquisition in which budgetary constraints necessitate reduced fielding of new systems and greater emphasis is placed on research and development. This program will pioneer the leveraging of R&D to upgrade, sustain and modernize existing fielded equipments in order to maintain the competitive technological advantage that the U.S. now possesses in defense capabilities.

3. LIGHTSAT COMMUNICATIONS TECHNOLOGY DEMONSTRATIONS

Another activity addressed by the ASTP is the actual on-orbit demonstration of complete systems. This provides "proof of the pudding" for all the preceding phases of R&D effort in systems studies, and subsystem and component development in the form of conclusive on-orbit operations.

3.1 MACSAT

The initial constellation of DARPA lightsats was launched on a Scout booster in May 1990, two years after startup of the program. It included two UHF store-and-forward satellites, the MACSATs. It was planned to demonstrate their operation to tactical commanders in as realistic environments as possible. The demonstrations would display global message relay for manpack terminals, to be interoperable with existing equipment.

The MACSAT schedule demonstrated an accelerated response and development time compared with more conventional system. The MACSATs performed communications operations in support of Operations Desert Shield and Desert Storm (for the U.S. Marine Corps), and were used in training demonstrations for the Army and the Navy and for Antarctic support for the National Science Foundation. Numerous demonstrations including transmission of digitized photographs have been accomplished. The physical characteristics of the MACSATs are as follows:

- 16-sided prismoid
- 24-inch diameter, 14-inch height
- 136 pounds
- Gravity gradient stabilized, assisted by magnetic Z-coil
- Semi-monocoque construction

The MACSATs are placed in a near polar circular orbit (89.9 degree inclination) at 370 nautical mile altitude. Up to 1,024 mailboxes may be accessed. The communications package operates with UHF frequency shift keying (FSK) receivers and either a 10-watt transmitter or a 60-watt high power transmitter. The standard data rate is 2.4 Kbps, although demonstrations at 4.8 Kbps have been accomplished. The electric power subsystem uses a body-mounted, 54-solar cell array and a 144-watt-hour commercial nickel cadmium (NiCd) battery pack, operating in an 18-volt DC bus.

3.2 Microsat

A second constellation of seven communication satellites, weighing 50 pounds each, were launched on a single Pegasus ALV on July 17, 1991. The satellites (known as Microsats) were designed to provide intra-theater voice or digital communications along with some store-and-forward digital data transfer. The electric power system included 18 solar panels with a 50-watt-hour commercial NiCd battery pack, a 5-volt linear regulator and a 5/15 volt DC/DC converter. The attitude control system included an earth sensor and magnetometers for reference, torque rods for spin-up and a 1.1-liter nitrogen tank for stationkeeping. The communications system allowed for either analog or digital communications. A digitally-controlled UHF FSK 10-watt transmitter and FSK receiver allowed voice communications or up to 4.8 Kbps digital data rate. Omni-directional blade antennas were deployed on both sides of the spacecraft.

Microsats had the following characteristics:

- 12-sided
- 19-inch diameter, 7.5 inch height
- Deployable antennas
- Spin stabilized
- Nitrogen cold gas propulsion

The Microsats were designed to be ejected from the carriage in such a manner as to place them in a single orbital plane with 400 nautical mile altitude circular orbits. Although a launch vehicle anomaly resulted in a much lower orbital altitude, the correct inclination and satellite spacing was achieved. A very successful demonstration program was completed before the satellites re-entered the atmosphere approximately six months after launch.

4. ADVANCED TECHNOLOGY DEMONSTRATIONS

Consistent with the new DoD Science and Technology Strategy, DARPA has initiated a number of new programs to quickly develop and apply advanced technologies into technology demonstrations. The purpose of these advanced technology demonstrations is to ensure we continue to pursue the advancement of enabling technologies that may be required in our future weapons systems and to reduce the risk for incorporating these technologies into future systems. These demonstrations include our existing Pegasus and Taurus launch vehicles and several new programs to demonstrate advanced satellite technologies for communications and remote sensing.

technologies into future systems. These demonstrations include our existing Pegasus and Taurus launch vehicles and several new programs to demonstrate advanced satellite technologies for communications and remote sensing.

4.1 Launch Vehicles

The launch vehicle portion of the ASTP is aimed at providing flexible and responsive capabilities to quickly insert payloads into orbit. Many of the current large vehicles are constrained to launches from either Vandenberg Air Force Base or Cape Kennedy. The primary objective is to demonstrate the capability to launch small satellites into space independently of the existing launch base infrastructure and to reduce the cost-per-pound to orbit for small launch vehicles that traditionally have suffered from the economy of scale barriers.

4.1.1 Pegasus

Pegasus is not a typical Government program, nor is it a commercial program. Rather, it is a dedicated team effort in which both the Government and industry cooperated to accomplish a demanding task in a very short amount of time with relatively little money. DARPA and the Air Force signed a transition agreement under which DARPA will turn over management for the remaining launches to the Air Force. This is a good example of DARPA's prime role in military R&D -- transitioning technology to the Services.

The Pegasus ALV successfully orbited payloads on its first flight on April 5, 1990, and on its second flight on July 17, 1991. This DARPA-sponsored program has demonstrated a responsive and economical capability for putting small payloads into orbit. This very successful program has met at least four major objectives:

- Validated the air launch concept (twice);
- Launched dissimilar NASA and Navy payloads on a single launch mission;
- Launched seven Microsats on a single launch mission; and
- Provided the first demonstration of a GPS receiver operating on a space launch vehicle during the boost phase ascent trajectory.

Following the first launch of Pegasus, some performance upgrades were incorporated for the second launch in July 1991. Although the second Pegasus launch experienced two significant inflight anomalies resulting in a lower-than-intended orbit injection altitude, the fact that it successfully orbited the seven Microsats is a visible measure of the robustness of its autonomous guidance and control system and ultimate capability. Design modifications to correct the problems have been developed and are in the process of being qualified.

4.1.2 Taurus

The Taurus SSLV will provide the capability to launch approximately five times the capacity of Pegasus from a ground-transportable system. It consists of a Pegasus without wings and a larger fairing sitting atop a Peacekeeper first stage. The entire launch system is transportable and can be operated from a bare base; launch site establishment is completed within five days and payloads can be launched within 72 hours of launch notification.

The current Taurus design can place approximately 2,000 pounds into a 400 nautical mile, 28.5 degree inclined orbit. It is expected that, with modifications (i.e., additional strap-on motors and a new apogee kick motor), Taurus will be able to place small satellites (600-pound class) into geostationary orbits.

4.2 Advanced Technology Standard Satellite Bus

The ATSSB project will design and develop a multimission-capable, small, standard spacecraft bus that incorporates a new approach to building satellites that can accommodate different types of "bolt-on" mission payloads. The bolt-on concept can best be described as a system having an isolated payload attached to a high performance, lightweight bus employing advanced technology through a single, standard interface (Figure 6). The ATSSB will be capable of accepting these bolt-on payloads from a variety of candidate mission areas including meteorology, communications, surveillance and tracking, target location and navigation.

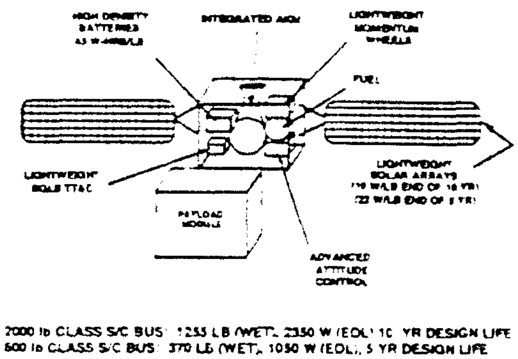

Figure 6. Generic Common Bus Configuration

Payloads employing the standard bus will be launchable from both small and large launch vehicles and capable of performing in a variety of orbits. The ATSSB project will provide designs for a versatile bus that will allow surge capability and ease in reconstitution of space assets.

The key features of the common bus are capability, affordability and flexibility. Capability comes mainly by employing newer technologies enabling greater performance in smaller packages. Autonomous orbit and attitude determination will enable satellite supervision, as well as satellite maintenance. The autonomous mission planning function will ultimately enable the support of tactical users' immediate tasking requirements. The affordability of all satellite systems is largely a function of total satellite (and associated booster) weight, as well as the number of units procured. Obviously, the non-recurring costs will decrease when each satellite ceases to be a special design case in itself. In addition, the flexibility of the bolt-on concept system can enable quick integration and change out, if necessary, immediately prior to launch.

The common bus provides a clean structural surface with a standard electrical interface to mate with a wide range of mission and payload types. A direct-load path is furnished for all payload elements and the bus, with provision for stacking payloads if multiple launches are desired.

DARPA has as an objective the increase in payload mass fraction from around 30 percent (the present level) to approximately twice that figure. This reversal in weight fraction invested in bus and payload, respectively, represents in itself a significant advance in space technology (see Figure 7).

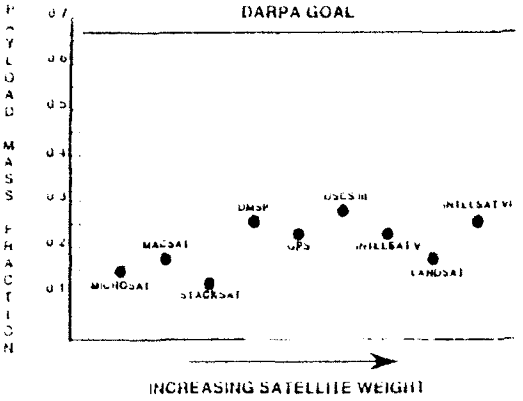

Figure 7. Payload Mass Fractions, Current Systems and Future Goals

The performance requirements for the ATSSB include: (1) support to payloads of up to 500 pounds requiring 400-600 watts of peak power, (2) three-axis attitude control, (3) standard communication links, (4) propulsion for orbit maintenance, and (5) a three-year mission life. The program will emphasize the use of common industry standards and will be scalable to support 1000-pound payloads with a seven-year mission life.

4.3 ASTEC Demonstration

This DARPA-led joint DoD program integrates the advanced EHF communications and satellite subsystem technologies under development in DARPA/DoD and flight demonstrates two experimental, lightweight, EHF small satellites. The goals of the program are to support the joint DoD Global Surveillance and Communication (GS&C) demonstration objectives:

- Develop, space-qualify and transition advanced technologies to support the mid-1990's MILSATCOM modernization decision milestones and integrated GS&C demonstrations;
- Assess the utility of small satellites to: (1) augment larger backbone satellites; (2) enable an evolutionary/affordable approach to MILSATCOM procurement/modernization; (3) provide quick reaction surge and specialized communications support; and
- Serve as an acquisition management testbed aimed at lowering costs and reducing "time to market" for new satellite systems.

The general program approach calls for the development, launch and demonstration of two technology satellites in geosynchronous orbits. Satellite 1 incorporates the prototype ATSSB and the Massachusetts Institute of Technology/Lincoln Laboratory-developed EHF payload containing very high data rate (VHDR: 274 Mbps, unprocessed EHF) and low data rate/medium data rate (LDR/MDR) capabilities. An integrated payload testbed will be built to support payload development, EHF MDR ground terminal development and on-orbit demonstration operations. The second satellite incorporates the ATSSB with an industrialized version of the Lincoln Laboratory LDR/MDR payload, augmented by multichannel MDR and an antenna nulling capability. Two satellites are required to accommodate the full set of technical objectives (including the industrialization of the EHF payload technologies) and the development time needed to mature the complex antenna nulling technology. The early launch of the first EHF payload will demonstrate the integrated technology in the timeframe needed to support integrated DoD demonstrations and MILSATCOM milestone decisions.

Figure 8 shows the functional layout of the core payload. The core of both payloads is an EHF communications package that provides 32 MIL-STD-1582C LDR channels and two MIL-STD-1810 MDR channels. The data rate for the LDR channels is 75-9600 bps and for the MDR channels, 4.8-1638.0 kbps. The core EHF communications package will provide a variable beamwidth antenna that uses a dichroic lens to support both uplink and downlink with a single reflector. Two earth coverage horns are included. The payloads will perform all signal processing required to maintain the MIL-STD-1582C and MIL-STD-1810 transmission formats. As a design goal, the payloads will have a modular design to allow early technology upgrades, as well as scalability, for increased channel capacity and extra spot beams.

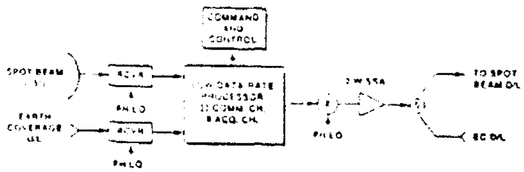

Figure 8. ASTEC EHF Payload

The technology advances gained as a result of the ASTEC technology demonstrations will greatly impact future MILSATCOM systems through the introduction of new capabilities, incorporation of new advanced technologies and demonstration of new implementation alternatives such as small satellite augmentation. The ASTEC payloads implement the following new MILSATCOM services:

- First on-orbit MIL-STD-1810 (EHF MDR) capability
  - Supports 4.8-1638.0 kbps data transmission
  - Includes four channels on Payload 1 and eight channels on Payload 2
- Unprocessed (i.e., transponder), VHDR EHF capability (Payload 1 only)
  - Supports scintillation-resistant delivery of wideband data to/from remote locations
  - Supports data rates up to 274 Mbps from a 10-foot, 500-watt transmit terminal, through the ASTEC VHDR transponder, to a 20-foot receive terminal

- Provides significant nulls over 2 GHz bandwidth
- Places null on jammer automatically

- Variable antenna spot beam capability (1.0-2.5 degrees)
  - Allows tailoring of the beam for various theater sizes
  - Allows tailoring of the beam to support various terminal requirements

- On-orbit payload reconfiguration capability
  - Allows re-allocation of beam/channel group pairings
  - Supports dynamic re-allocation of resources to support changing networks

The ASTEC payloads implement these services by employing advanced technologies in key subsystem applications that will reduce payload weight and power consumption while increasing capability. As a direct comparison, the ASTEC MIL-STD-1582C 36-channel group achieves a 65 percent reduction in weight and a 50 percent reduction in power over the 36-channel group implemented on the Fleet Satellite Communications (FLTSATCOM) EHF Package. The technologies that enable this reduction include application-specific integrated circuits for signal processing; direct digital synthesis of the local oscillator frequencies; lightweight, flangeless waveguide for the beamforming networks; and either permeable base transistors or high-electron-mobility transistors for high efficiency, high power solid-state power amplifiers.

4.4 CAMEO Demonstration

The CAMEO (Collaboration on Advanced Multispectral Earth Observation) Program will integrate advanced optical and sensor technologies with the ATSSB to demonstrate and space-qualify a new, lightweight, multispectral, remote sensing satellite. The program offers an important two-fold opportunity: to demonstrate technology that simultaneously addresses critical DoD and civil needs, and to advance the concept of using small satellites for rapid, affordable capability insertion beyond DoD into the civil/national space arena. The goals of the program are to demonstrate:

- "Dual use," multipurpose, advanced, remote sensing technologies for: (1) DoD-Wide Area Surveillance, (2) Civil-Global Climate Research and (3) Civil-Environmental Monitoring;

- The use of small satellites to rapidly augment on-orbit capability and affordably modernize both DoD and civil remote sensing satellite constellations; and

- New payload operations concepts and the direct downlinking of usable data to DoD and civil consumers.

The payload concept is distinctly different from typical "dual use" approaches. Rather than develop a sensor that can be used only for a single purpose by "dual users" (i.e., cloud imaging for DoD and civil needs), this program will develop advanced sensor technology adaptable to *multiple* purposes by *multiple* users.

Specific technical objectives are to:

- Develop and space-qualify an advanced, multispectral, remote sensing payload for small satellites;

- Demonstrate multimission usage of ATSSB for remote sensing payloads in low earth orbit; and

- Develop and demonstrate Common Data Link (CDL) standard technology for small satellites and direct downlink connectivity to mobile earth terminals.

Our first activity was to identify the DoD and civil remote sensing requirements and needs that could be met best by a multispectral payload. Existing DoD requirements and measurement needs associated with civil programs such as Landsat, NASA's Earth Observing System (EOS) and the Strategic Environmental Research and Development Program were surveyed, from which emerged clear areas of overlap in spectral coverage, spectral resolution, etc., that indicate many requirements could be addressed using an integrated multispectral payload. The requirements framework was used to define the general characteristics of a payload and an in-house study was conducted to define a point design. This design proved feasible for implementation as a small satellite payload on the ATSSB. The current program plan calls for system development consistent with an FY96 launch.

The first area surveyed was climate research. The highest priority measurements for detecting and characterizing the global climate change signal and the pattern of this change have been well articulated by the U.S. Global Change Research Program and many individual scientists such as Dr. Jim Hansen, Director of NASA's Goddard Institute for Space Studies. The core measurement requirement essential to characterizing the climate system is long-term monitoring of the earth's radiation budget. These measurements have not been made for approximately two years and none are planned before the deployment of a radiation budget radiometer on a Japanese satellite in the mid-1990s and the first EOS system in the late 1990s. It is essential to understand man's impact on the climate system, anthropogenic climate forcings. Space measurements of aerosols, ozone, water vapor and surface reflectivity (albedo) are required, all of which, at present, are measured inadequately or not measured at all. Lastly, the climate feedback mechanisms are poorly understood and modeled. It is essential to understand whether man's impact on the climate system results in large scale changes (e.g., cloud cover) that positively or negatively reinforce the trend toward global warming. The most important of these requires detailed global measurements of cloud phase and particle size.

The second measurement area considered is environmental quality monitoring, including parameters such as air and water pollution, hazardous waste site monitoring, wetlands and land use monitoring and surveillance of natural and man-made disasters for emergency response. Most of these parameters have signature features that emerge from multispectral measurements. Although Landsat and Systeme Probatoire d'Observation de la Terre (SPOT) have addressed many of these, their spectral bands were not optimized to measure them and their spatial resolution is inadequate.

The third area surveyed was DoD's need for multispectral imagery. The primary mission considered is battlefield situation monitoring, which includes area delimitation and target nomination. Area delimitation is that process by which sensor data is used to exclude large regions that otherwise require wasteful, repetitive searches using high value assets. For example, terrain relief and trafficability maps can be used to delimit regions where transportable vehicles can be employed. Target nomination includes identifying localized regions of interest for cueing of higher resolution assets. Detection of large scale camouflage deployment could be used as a target nomination metric.

Other primary applications include support to the identification and recognition process as well as targeting. In the latter, the roles envisioned for a multispectral system include 3-D terrain mapping and weather updating. Ancillary DoD missions include monitoring the growth and transportation of illegal crops and drugs, coastal zone monitoring including bathymetry, and large-scale change detection (runway extension, etc.).

The overlapping requirements for spectral coverage and resolution, number of spectral bands, spatial resolution, FOV, calibration and radiometric sensitivity required to address these three mission areas have been identified. The striking similarity is that (with the exception of one aspect of climate monitoring) almost all measurement categories require a few to several bands between 0.4 and 5.0 microns. The total number of unique bands required exceeds 25. For both DoD and environmental quality monitoring, moderate to high spatial resolution is required (5-20 meters) over moderate sized FOVs (10-50 km). All climate measurement parameters, however, require a WFOV at relatively coarse resolution (3-10 km).

This analysis has led us to define an integrated payload consisting of three sensors: a narrow FOV (NFOV) sensor to address most DoD and environmental quality monitoring measurement needs, a WFOV sensor to address most climate monitoring needs and DoD's battlefield weather requirements, and a radiation budget radiometer to measure the earth's total outgoing radiation.

Elements of the integrated system include the space segment, bus control element and payload ground station. As described earlier, the ATSSB will be used. It is capable of a 50 percent mass fraction, for advanced technology "bolt-on" payloads, and will use CDL as the communication system. The integrated payload will consist of the above three sensors. Satellite bus control will be managed through the Air Force Satellite Control Network (AFSCN). Payload tasking, sensor data downlink and data exploitation will occur at the payload ground station, which will include the existing Mobile Interoperable Surface Terminal (MIST) and a small, powerful processing system. All sensor data will be transmitted for further processing and archival through the payload ground station.

The integrated payload design approach includes several essential features. First, the spacecraft is sized for launch on the Taurus SSLV using the ATSSB. Total weight and peak power budgets of 900 pounds and 600 watts, respectively, have been established. The system design life is three years, which will allow system designers to balance performance with reliability to minimize overall system cost. CDL has been adopted to facilitate the downlink of high data rate (HDR) NFOV imagery in a compact, power efficient, affordable system.

The NFOV instrument (Lightweight Multispectral Imaging Sensor, (LMIS)) will provide 5-20 meter resolution with spectral bands covering the 0.4-5.0 micron range. The current baseline design for LMIS includes 4-6 bands in the visible and near infrared (0.4-1.0 μM) and shortwave infrared (1.0-2.5 μM), respectively, and 2-3 bands in the medium wave infrared (3.0-5.0 μM). Multilinear arrays with striped spectral filters will be used and with the sensor operated in a pushbroom mode. The total FOV is expected to be about 15 km.

A portion of the FOV will be used to demonstrate a hyperspectral imaging capability using innovative filter technology. A candidate is the linear wedge filter which, when integrated with a focal plane array, provides a continuum of spectral bands at high resolution (10-20 nM). LMIS will include on-chip spectral and spatial aggregation modes. This feature, combined with onboard image bandwidth compression, provides a means for transmitting all LMIS imagery within the downlink capacity.

The WFOV instrument (Multispectral Pushbroom Imaging Radiometer, (MPIR)) will have 10-15 bands covering the visible to longwave infrared collected at 1 3 km resolution. The MPIR FOV will be approximately 1000 km, achieved by several small, very WFOV optical systems operating in the pushbroom mode.

Finally, the radiation budget radiometer most likely will be an improved version of flight-proven instruments flown by NASA. It measures total outgoing shortwave and longwave flux over a horizon-to-horizon FOV.

The measurement capability of this integrated payload distinguishes it from existing and planned remote sensing systems. MPIR has coverage and resolution similar to the Defense Meteorological Satellite Program and the TIROS series, but its spectral bands have been chosen to measure quantities not available from these systems. Examples include total ozone, aerosol loading, surface albedo and cloud phase and particle size. LMIS also has characteristics that render it fundamentally distinct from Landsat, SPOT and the High Resolution Imaging Spectrometer (HIRIS) system planned for EOS. The number of bands and spatial resolution provide an ability to measure quantities well beyond the limits of Landsat and SPOT. These include rock and mineral type and water pollution and wetland monitoring for the civil community as well as mapping, trafficability and battlefield monitoring for DoD. Finally, the spatial resolution of this system will support detection of camouflage and other military equipment in many deployment scenarios. The spatial resolution of HIRIS, a factor of 2-4 coarser than this system, is inadequate for these missions. Finally, the radiation budget radiometer flown on this satellite will fill the measurement gap prior to EOS deployment.

Successful completion of the CAMEO program will yield a number of diverse technology and concept breakthroughs critical to future augmentation and modernization of DoD and civil remote sensing architectures. The development and spaceflight of a lightweight, multispectral imager will demonstrate the applicability of advanced focal plane and optical technologies on a small satellite. LMIS will operate at raw HDR, requiring the use of high-speed analog and digital processing electronics and bandwidth compression chips. This will be the first demonstration of these technologies on a small satellite.

The CAMEO program will demonstrate the benefits of the ATSSB. The opportunity will exist in this program to further examine and highlight the significant acquisition time and cost reductions possible through the introduction of standardized/simplified interfaces and the elimination of non-recurring engineering. The weight of the multispectral payload is expected to fully exploit and test the capability of the bus to meet the 50 percent payload mass fraction goal. This represents nearly a factor of 2 mass fraction improvement over current satellites.

Finally, the lightweight implementation and space-qualification of the CDL technology (now, the airborne surveillance datalink standard) will enable transition of the CDL standard into satellite platforms. This is an extremely important technology progression toward the far-term DoD goal of global C^3I connectivity among all tactical and strategic platforms.

5. SUMMARY

The DARPA ASTP is aggressively supporting R&D efforts aimed at enabling technologies for new systems concepts and advancing space technology state-of-the-art. These new technologies will be demonstrated in prototype or breadboard systems and transferred as rapidly as possible into operational systems. In fact, the primary objectives of DARPA's ASTP are to *transition* advanced technologies, operational concepts and acquisition approaches to the appropriate DoD organizations and to *transfer* these same technologies to industry in support of future DoD procurements. Already, the successful launch and operation of the Pegasus ALV, MACSATs and Microsats have been carried out. Recent demonstrations of MACSAT's store-and-forward communications capabilities have been demonstrated to the Services in the field and preliminary responses from operating units are very favorable. For example, the Navy is pursuing its Arctic Communications Program based on MACSAT systems concepts and technologies. The Air Force, NASA, SDIO and civil customers have awarded contracts for Pegasus launches. DARPA will continue to pursue promising areas for achieving new capabilities with smaller and lighter advanced technologies and pave the way for showing how development of space systems can be accomplished more quickly and at greatly reduced cost.

Discussion

Question: You mentioned an IPSRU program where platform jitter is assessed/eliminated using a laser. May I ask:

a. If the intent of the program to measure jitter (for later post-processing), or to eliminate/compensate for jitter through a slaving control mechanism?

b. Is literature is available on this matter (open or at least releasable to NATO)?

Reply: a) The unit alone provides very precise and accurate line-of-sight offset measurements for high frequency, low magnitude jitter. When used in concert with a fast steering mirror, or other closed loop system, it can actively remove jitter.

b) The organization developing IPSRU is Draper Laboratory in Massachusetts. The program manager is Mr. Jerry Gilmore.

Question: Would you comment on the status of CAMEO?

Reply: The CAMEO, ASTEC and ATSSB (standard bus) funding has been included in the President's Budget submission to Congress in Fiscal Year 1993 and is in the Five Year Defense Plan. The initial technology work for EHF Comms has been conducted for DARPA by Lincoln Laboratory and awaits further funding. CAMEO payload specifications have been developed. The ATSSB is currently in source selection. The Congress deferred these programs in FY 93, stating their opinion that we are premature in starting the program. We plan to re-plan the program for a full start in 1994.

Question: Referring to submarine laser communications, where you indicated that the submarine would uplink a signal for geolocation purposes to assist in downlink laser aiming, what measures do you propose for preventing exploitation of this uplink radiation by hostile forces for the same purposes?

Reply: There will be, in general, sufficient geographical separation that exploitation will not be possible. Use of EHF ensures narrow beamwidths.

SAR SENSORS ON TACSATS:
A FEASIBILITY ASSESSMENT

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SUMMARY

The feasibility of installing SARs on board lightsats for tactical and strategic observation missions is discussed, emphasizing high resolution and short revisit intervals as main system drivers. Lightsat constellations design criteria are presented for both global and limited latitude belt coverage. Pros and cons of sunsynchronous versus medium inclination non sunsynchronous orbits are discussed.

Gross system trade offs for the SAR sensor are then addressed, in the specific context of a resource-limited lightsat, stressing the achievement of resolutions better than 5 m, swaths greater than 20 Km, access angles of at least 30°, small antenna dimensions and a reasonable power consumption. Both X and C band SAR solutions are outlined. Data transmission alternatives are also discussed, in the context of tactical and strategic scenarios, outlining their projected performance.

Constellation orbits control and platform attitude are also addressed for their impact on mission, SAR image quality and satellite design requirements. Key aspects of critical platform subsystems are also identified.

1. INTRODUCTION

Lightsat constellations are receiving considerable attention by industry, governmental Agencies and commercial carriers. Their potential for strategic communications is well understood by military planners, which contributed to the fast spinoff of major projects being undertaken by some American system houses. Less well perceived are the lightsats capabilities for remote sensing and observation tasks for both civil and defense uses. In 1990, anticipating the need for new and unconventional solutions to observation satellites, Alenia Spazio started an internal study addressing the feasibility of small SAR satellites, stressing the achievement of high resolutions and short revisit intervals. The '91 Gulf War provided further stimuli in that direction, reinforced by the world-wide trend concerning space and defence budget allocations. First results were published in [1]. Since then, further work on system architectures and trade offs has been performed, concentrating on military applications and the near-term feasibility with available technologies, thus avoiding long and costly new developments. This paper reviews some of the most recent results achieved so far.

2. SAR OBSERVATION MISSIONS

SAR sensors operational value comes from their independence from time of the day and clouds cover. Achievable ground resolutions, swaths and access angles are comparable to those of panchromatic optical sensors. Nevertheless SARs respond differently to the physical features of the observed scene: this may be exploited to enhance certain target characteristics. SAR and optical satellites can therefore complement each other.

However, in a tactical scenario, where weather and time-of-day independence of the observations are a premium factor, SAR satellites outperform the optical ones from an operational viewpoint.

In addition, constellations of SAR satellites can offer performance unmatched by larger spacecraft flown individually, such as inherent redundancy, more frequent revisits and an easier access to in-orbit resources. If SAR sensors can be flown on lightsats the total system cost can be considerably reduced, while achieving more flexibility in deploying and managing them.

2.1. System configuration alternatives

We will discuss representative missions stressing the hi-resolution ones for defense applications. SAR lightsats can be launched on-demand, for short duration observation tasks over specific areas, in the event of geo-political or military crises. Alternatively, long duration missions with lightsats constellations can be envisaged to perform continuous observations within a given latitude belt.

The cost-effectiveness of short duration missions is nevertheless questionable, considering the need for a permanent ground infrastructure which has to be operated and maintained anyway, and the satellite and launch costs which are poorly amortized. Permanent constellations can, instead, offer even better global performance at a higher initial cost which is however amortized over a much longer service period. Furthermore, a permanent constellation can provide shorter revisit intervals and, outside crises periods, strategic observation services as well as remote sensing functions for government and public use, with specific reference to post-disaster damage assessment. We will thus concentrate, in the following, on lightsat constellations.

2.2. Requirements overview

An increasingly important requirement concerns the revisit interval with which military relevant sites must be observed: typical values from few days to 24 hours apply mainly to fixed assets. Observing targets variable in time or space requires, however, much shorter revisit intervals, of the order of few hours. Concerning ground resolution, 5 m are sufficient for detection, and in some cases recognition, of strategically relevant fixed or low-mobility assets. Tactical applications need, however, better resolutions down to 2 m or less.

The swath width , or instantaneous field of view, must be commensurate to the theater: 20 to 40 Km , depending from the ground resolution, are likely values. The swath must also be electronically repositioned inside the access angle, providing a high operational flexibility in gathering SAR images of selected spots during satellites' overpasses. Wide access angles are instrumental to secure a coverage without 'holes' and to minimize the constellation's satellites number.

Time delay minimization between a request for data and its availability to the end-user is of utmost importance, specially in a tactical situation. Direct access to the satellites, during sites overpasses, followed by local data processing and interpretation by multiple data stations deployed in the theater, are seen as an effective answer to such needs.

In a strategic scenario there is the additional requirement of being able to observe distant sites with a short turnaround time. Relaying imagery data via a Data Relay Satellite network is the most logical and performant solution to the problem.

3. CONSTELLATIONS DESIGN CRITERIA

We must distinguish between systems aimed at providing a global Earth coverage and those intended to cover a narrower latitude belt around the equator.

As a matter of fact, most Countries where political instability is expected to occur also in future are included in the 50° N to 50° S latitude belt: which may justify tailored constellations.

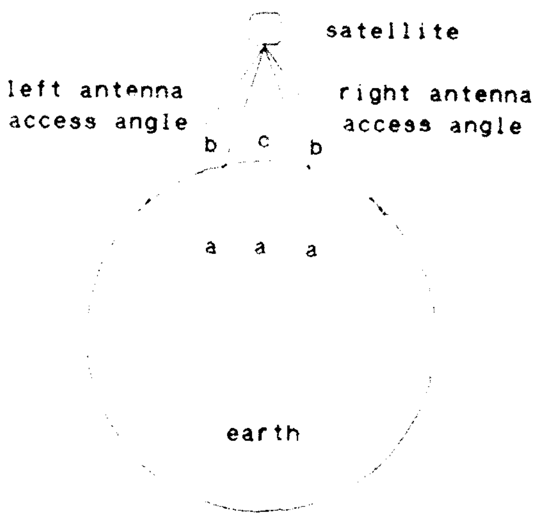
3.1. Constellations for global coverage

A modular solution consists in injecting N equispaced lightsats in single plane sun-synchronous orbits; the SAR can be provided with just one or two antennas looking on both sides of the flight path. Multiple equispaced orbital planes can cope with very stringent revisit interval requirements, i.e. less than 12 hours.

3.1.1. Lightsat Constellations with one-antenna SARs

The capability of a SAR to image target areas inside a wide earth strip is proportional to the access angle, defined by the minimum and maximum SAR off-nadir angles. The constellation design criteria should provide for a contiguous earth coverage, within the above constraints. To this end, the fundamental interval is divided into N subintervals whose width corresponds to the SAR antenna access angle projected onto the equator. If the minimum and maximum off-nadir angles satisfy the constraints of Fig. 3.1.1., then the strips accessible to the N satellites will be contiguous.

Fig. 3.1.1. SYSTEM GEOMETRY



In one orbit period, the N satellites will have covered an Earth slice as wide as the fundamental interval, and the cycle will repeat providing a continuous Earth coverage up to a latitude close to the sunsynchronous orbit inclination.

With the constraint of keeping the minimum and maximum off-nadir angles close to 20° and 50° respectively, the number of satellites N depends from the orbit altitude and access angle, as shown in Fig.3.1.2. and Table 3.1.1.

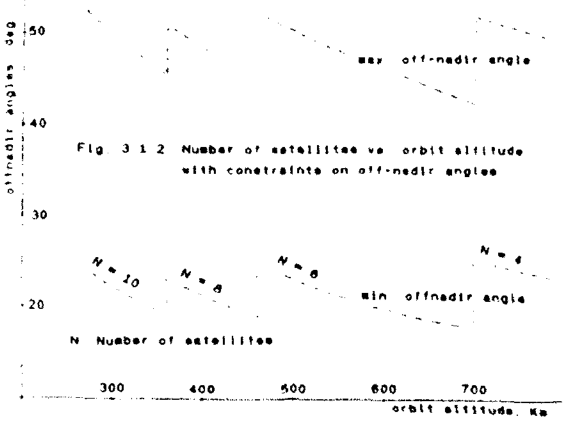


Fig 3 1 2 Number of satellites vs orbit altitude with constraints on off-nadir angles

orbit height(Km)	N° satell. per plane	Delay between two S/C passes(*)
270-360	10	18.3-20.5 min
360-460	8	22.9-23.4 "
460-700	6	31.2-32.6 "
> 700	4	> 49.5

(*) for two-antenna SARs

Table 3.1.1. Satellites number vs. orbit altitude

With this arrangement, the concept of repeat cycle looses partly its meaning: in other words since a contiguous earth coverage can be achieved dayly and SARs are not sensitive to sun illumination conditions, orbit repeat cycles of 1,2,3 or more days can be chosen. This provides more freedom in choosing the orbit altitude, which is a critical factor for SAR dimensioning due to lightsats power limitations. On the other hand, the variability of the incidence angle, with which a site can be observed during subsequent days in the repeat cycle, may be considered a positive feature in view of the possibility of implementing a slow sampling rate incidence angle diversity system, which may enhance the recognition of fixed assets.

Since SARs can operate day and night, the one-orbit plane configuration achieves a nominal 12 hours revisit interval over most sites. If revisit intervals shorter than 12 hours are required, M equispaced orbit planes can be implemented, and the average revisit interval will be 12/M hours. The price to be paid is an M-fold increase in the satellites number.

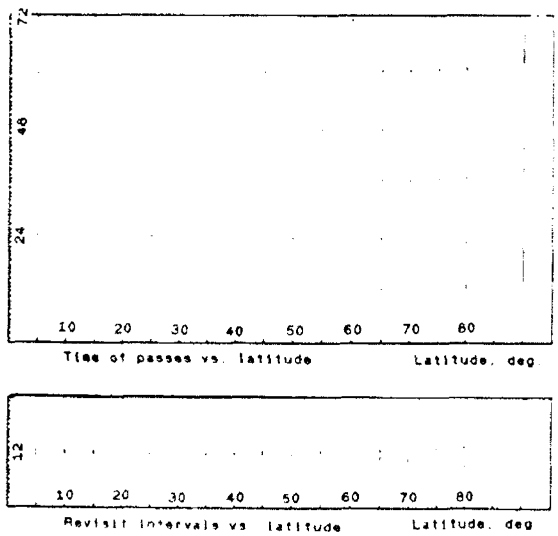


Fig. 3.1.3 8-satellites single plane constellation Sunsynchronous orbit, 362 Km altitude; right antenna

The constellation design criteria were verified through extensive computer simulations. Results for an 8 equispaced, single plane, satellite constellation in a 362 Km sun-synchronous orbit are shown in Fig.3.1.3. for a SAR antenna looking to the right of the flight direction.

Installing the SAR antenna on the left side decreases slightly the maximum observable latitudes, a feature common to all retrograde orbits. Fig. 3.1.3. also shows the revisit intervals vs. latitude: the 12 hours value is confirmed, with minor deviations above 60° due to the inclination of the orbit plane combined with the orientation of the SAR antenna.

3.1.2 Lightsat Constellations with two-antennas SARs

The system geometry in Fig. 3.1.1. is such that the access angles, and the gap in between, subtend identical arcs on the equator, corresponding to 1/Nth of the fundamental interval. After 1/Nth of the orbit period, the previously unaccessible strip is now visible by the 2nd satellite; after 2/Nth of the orbit period, the strip accessed by the left antenna on the 1st satellite is now visible by the right antenna on the third satellite. Each strip can be, thus, accessed twice by different satellites with a time delay of 2/N of an orbit period: see Table 3.1.1. In summary, the presence of two antennas does not change the number of satellites required to achieve a contiguous Earth coverage, but allows looking at the same target with different incidence angles after a rather short time delay. This allows implementing a fast sampling rate incidence angle diversity system, for target detection and recognition enhancement purposes. Besides, target shadowing effects will be lessened, since the same area can be observed from both right and left directions. With a two-antenna SAR payload, stereo SAR imaging in lateral vision could be ultimately implemented, provided that suitable processing techniques will be developed.

3.2. Constellations for reduced latitude belt coverage

Observation systems for limited latitude belt coverage exploit the characteristics of low to medium inclination non sun-synchronous orbits. This is possible due to the SAR indipendence from sun illumination conditions. Low orbit inclinations give several advantages.

First, the number of satellites required to contiguously cover the fundamental interval is less than with sun-synchronous orbits. This is shown in Table 3.2.1.

repeat/ orbits	inclin. (°)	height (Km)	N° Sat. per plane
1/15	30	482.1	4
" "	40	488.4	4
" "	50	496	5
2/31	30	328.4	5
" "	40	334.4	6
" "	50	342.7	7
3/46	30	378.9	4
" "	40	384.9	5
" "	50	392.9	6

Table 3.2.1. Non sunsynchronous orbits constellations: satellites vs. altitude and inclination

giving the number of satellites per plane vs. the orbit inclination and altitude, with the additional constraint of minimum and maximum off-nadir angles of 20° and 50°.

Second, at latitudes close to the orbit inclination the ground tracks are denser and move east-west instead of north-south. This feature may be very usefull in certain regional scenarios.

However, if all satellites are in the same plane there will be a clustering of revisits every 24 hours. To achieve an even spreading of the revisits trough the day, it is essential to redistribute the satellites in M orbital planes, with a proper phasing (right ascension and satellites' true anomaly). An example of application was presented in [1] and is also mentioned in a companion paper [2].

Third, one may combine orbital planes having different inclinations and same fundamental. Coverage continuity needs a proper phasing of satellites orbital parameters, achieving also a reduction of the revisit intervals at latitudes corresponding to each orbit plane inclination. This approach reduces the spread in the average revisit intervals, typical of constellations with one orbit plane inclination only.

3.3. Impact of orbit plane orientation on lightsats design

The orbit plane orientation plays a fundamental role in lightsat design. Single plane down-dusk sunsynchronous orbits are very convenient for SAR satellites: infact fixed solar arrays may be used since they are always illuminated by the sun. This facilitates continuous SAR operation, unrestrained by batteries capacity. Solar array wings are parallel to the orbit plane minimizing the drag effect. One side of the spacecraft is always exposed to cold space providing an ideal heat sink for thermal control.

Multiple planes sunsynchronous orbits can be oriented symmetrically w.r.t. the 6 AM-6 PM plane implying sun-tracking solar arrays. Earth shadowing for about 50% of the orbit will, anyway, result requiring to support the SAR operation from on-board batteries. Besides, the variable sun vector incidence on the satellite, during the orbit, complicates the thermal control. Nevertheless the sunsynchronicity and system symmetry , the latter only in case of an even number of orbit planes, will help in controlling the growth in system complexity.

This will not be so in case of non-sunsynchronous inclined orbits. The dayly nodal shift will cause a slow motion of the orbit plane w.r.t. the sun vector. The system design must, then, cope with orbital period time-varying phenomena as well as with slowly changing ones.

This may considerably affect the satellite design, SAR operation and mission planning. With medium inclination orbits, the solar array design becomes even more critical, requiring a 2-DOF sun-tracking mechanism. The necessity for continuous solar wings reorientation will cause a time-changing satellite cross-section , adding another variable to the problem of drag compensation. Besides, variable external torques may impact the satellite attitude control, specially at low altitudes. The thermal control also becomes more critical due to the full variability of environmental conditions.

In summary the complexity and cost of SAR lightsats increases going from down-dusk to multiple planes sunsynchronous orbits and, eventually, to single or multiple inclined non sunsynchronous orbits. The increased satellite complexity necesarily reduces the payload accomodation capability for the same launch mass. These considerations must be borne in mind when evaluating the mission benefits of various orbit alternatives.

4. SAR-CARRYING LIGHTSATS

4.1. Capabilities and limitations of lightsats

The design of a SAR-carrying lightsat must follow a bottom-up approach starting from a set of constraints and defining which performance can be reasonably achieved. A lightsat carrying a SAR sensor for professional uses cannot be too small: a 500 to 800 Kg launch mass range was chosen, being within the injection capabilities, in LEO, of several planned small launch vehicles. For a 5 years lifetime, drag compensation is the dominant factor in sizing the propulsion system. A companion paper [3] shows that, for long mission durations, electric propulsion is mandatory to keep the propellant mass within 100 Kg at very low altitudes. The associated DC power consumption must be considered in satellite power plant sizing.

The SAR antenna cannot be too large: when folded and stowed it must fit the limited volume inside the shroud envelope: its mass and area must be compatible with the attitude control capabilities and should not contribute significantly to drag. An antenna lenght of 6 m and a width of 1.5 m were defined as upper bounds.

The SAR will normally operate for a fraction of the orbit period, so that two parameters are of concern: the average energy per orbit and the required peak power. Both increase with orbit height, therefore SAR-carrying lightsats must preferably fly rather low.

The average energy collected by the lightsat depends from the chosen orbit plane orientation w.r.t the sun. Typically, it can be in the 0.7 to 1.8 KWh/orbit range. Considering the energy consumed by essential platform functions, that available to payload is between 0.5 and 1.3 KWh/orbit.

For a typical 20% operating duty, a SAR DC peak power consumption of 1.5 to 4 KW could be fitted from pure energy considerations. Nevertheless, this would imply to rely heavily on batteries. More conservatively, a DC power to SAR allocation in the 0.5 to 1.5 KW range was retained for system trade-offs. Concerning data rates, an upper technology bound of 200 Mbit/sec. was also assumed. In view of the above, a SAR payload mass allocation in the 150 to 250 Kg range results, for a 500 to 800 Kg spacecraft launch mass range.

4.2. SAR SENSORS FOR LIGHTSATS

We review the main trade-offs impacting the SAR design in presence of constraints. The SAR will normally operate in the STRIPMAP mode: the swath will be electronically repositioned, inside the access angle, by beam steering in the elevation plane only. This operating mode was found to be adequate for the intended high resolution missions, given the lightsats constraints. Other SAR operating modes, such as SCANSAR, are also available if required by the mission.

4.2.1 Image interpretation, target characteristics, and S/N

For extended targets high resolution images interpretation is more related to pixel size than to radiometric resolution. A single look S/N of 5 dB was chosen for SAR dimensioning. SAR operating at different frequencies respond differently to the physical characteristics of the Earth surface, in terms of backscattering coefficient vs. the incidence angle. For the same backscatter coefficients and swath width the average transmitted RF power increases fourfold from S to X band. Nevertheless the average backscatter from typical ground surfaces also increase by 6 dB from S to X band, so that the two effects compensate and it could be possible, in principle, to transmit the same power at both bands. For discrete targets, however, the determining factor is the contrast ratio against the speckled clutter The relative merits of X vs. S band are still unresolved due to the large variety of possible scenarios and the scarcity of experimental data at X band. Assumimg frequency independent target Radar Cross Sections, the SAR should be designed for the same S/N indipendently from frequency. This criterion results in an increase of the transmitted RF power with the operating frequency, penalizing the X-band choice. In the following reference is made to a sigma-nought of -15 dB independent from frequency and off-nadir angles.

4.2.2. Frequency choice, access angle, and resolution

In sizing the SAR access angle, too low off-nadir angle values should be avoided, not to look at targets with near vertical incidence. Too large values should also be avoided to reduce shadowing effects and excessive image distorsions.

Besides these qualitative considerations more precise limits are set by both regulatory and lightsats accommodation restraints. The upper value impacts the transverse antenna dimensions, as shown in Fig.4.2.1. showing the antenna width vs. the maximum off-nadir angle, operating frequency and spacecraft altitude to swath ratio H/Sw.

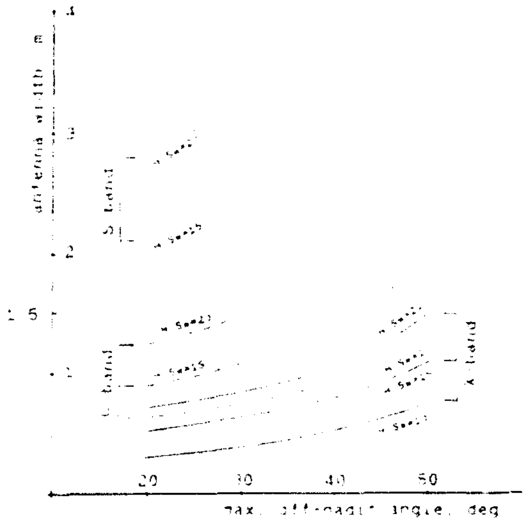


Fig. 4.2.1. Antenna width vs. frequency band, maximum off-nadir angle and H/Sw ratio.

This Figure shows that, at X-band, a 50 off-nadir angle is both feasible and compatible with the assumed antenna constraints even increasing the H/Sw ratio to 20 (e.g.: H=500 Km and Sw=25 Km).

At C-band, one has to superiorly limit the maximum off-nadir angle to about 40°, loosing in coverage; alternatively a maximum H/Sw ratio of about 12 could be chosen implying however an altitude of only 300 Km for a 25 Km swath.

S-band antennas can hardly be accomodated on lightsats at all off-nadir angles and orbit heights.

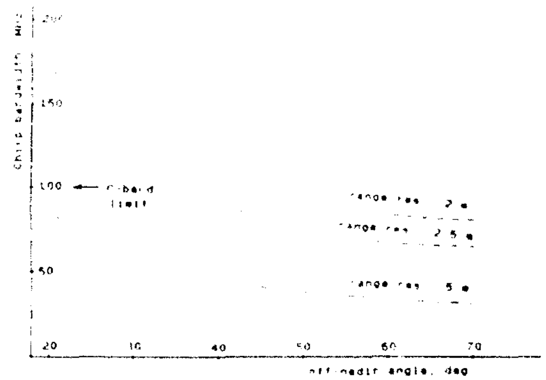


Fig. 4.2.2. Chirp bandwidth vs. range resolution and off-nadir angle.

The lower bound of the off-nadir angle is determined by the SAR chirp bandwidth, as shown in Fig.4.2.2. for range resolutions between 2 and 5 m.

The instantaneous SAR bandwidth must be compatible with international bandwidth allocations for Radar systems operating in the various frequency bands. It turns out that, considering the 100 Mhz bandwidth allocation for C-band space radars, a 5 m. range resolution limit must be accepted, or a severe restriction in the minimum offnadir angle to about 40° would result, impacting negatively the coverage or the number of satellites in the constellation. On the other hand, a 2 m range resolution at 20° minimum offnadir angle is compatible with both X and S-band choices, since bandwiths of respectively 300 and 200 MHz are available for this service.

Azimuth resolution, in the SAR stripmap mode, depends from antenna lenght and looks number. It is between 2 and 3 m for antenna lenghts in the 4-6 m range, with 1 look. The situation is summarized in Table 4.2.1. which identifies two possible system alternatives:
- a medium-high resolution C-band SAR for constellations in low to medium altitude orbits;
- a high resolution X-band SAR, for constellations in medium altitude orbits.

The S-band alternative is rejected being incompatible with typical lightsat accommodation constraints.

Freq. band	Avail. BW (MHz)	Range resol. (m)	Azim. resol. (m)	Maximum H/Sw (*)
S	200	2	2-3	< 20
C	<100	5	2-3	< 12
X	300	2	2-3	not compatib.

(*) at 50° off-nadir, Nlooks=1

Table 4.2.1. Impact of frequency band and lightsats constraints on SAR feasibility

4.2.3 X-band SAR: Swath, data rates, and RF power

This section focuses on X band SAR trade-offs. The swath, off-nadir angles, resolution, pulse lenght, orbit height, and average RF power are closely interrelated.

At a medium altitude of 360 Km and maximum off-nadir angle of 50°, achievable swaths and average RF powers vs. ground resolution and pulse lenght are plotted in Fig. 4.2.3. Increasing the pulse lenght one looses in swath width but peak powers decrease too. More specifically for pulse lenght greater than 20 microsec. peak powers below 1 KW are feasible, while below 15 microseconds multikilowatt peak power levels result, impacting HPA technology choice and reliability.

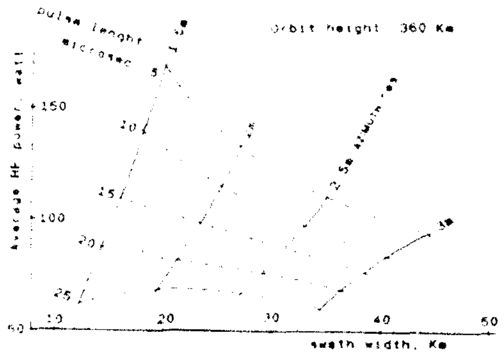


Fig. 4.2.3. Average transmitted RF power vs. swath width, azimuth resolution and pulse lenght.

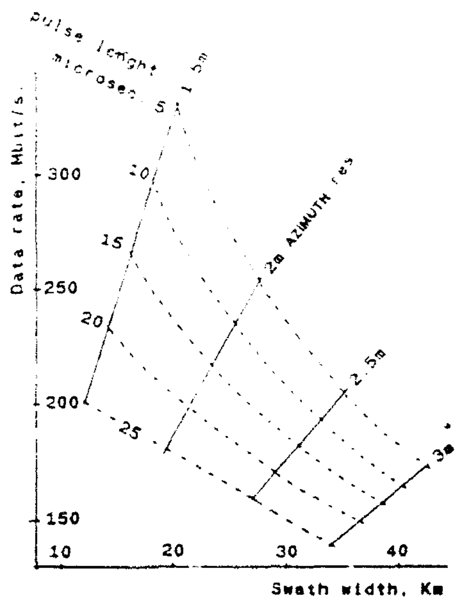


Fig. 4.2.4. Data rate vs. swath width, azimuth resolution and pulse lenght. Orbit height: 360 Km; off-nadir angle: 50°

Fig. 4.2.4. shows how the data rate varies with swath and ground resolution, assuming to transmit 4 bits/sample, ... is felt to be adequate for most tactical SAR imaging applications. The data rate increases with swath and even more rapidly with ground resolution, and is therefore a limiting factor for such satellite systems. The assumed 200 Mbit/sec upper bound is compatible with a 20 Km swath at 2 m resolution, or a 35 Km swath at 2.5 m resolution. Lower data rates would imply narrower swaths, and viceversa.

The average RF power increases also with orbit altitude, see Table 4.2.2.

Orbit alt. (Km)	Swath width(Km)/resolution(m) 40/3	30/2.5	25/2.5
280	65.8	63.6	44.1
360	84.6	81.8	56.7
460	108	104	72.5
560	127	123	88.3
660	157	143	101

Table 4.2.2. Average transmitted RF power vs. orbit altitude, swath and resolution at 50° off-nadir.

This confirms the desire for flying SARs at rather low altitudes, compatibly with coverage and revisits goals. Once the swath is chosen it is kept constant, independently from its repositioning inside the access angle, and so is the peak power for obvious hardware constraints. However, from 50° to 20° off-nadir, the antenna beam in the elevation plane must be broadened to cover the swath. As a result the pulse duration must be gradually increased to recover the beambroadening losses. Thus, at 20° off-nadir, the average transmitted RF power as well as the absorbed DC power, is about 1.5 times that at 50° off-nadir.

Other SAR modes can be implemented according to mission needs, trading off geometric with radiometric resolutions (multi-look modes), or azimuth resolution with swath widths (SCANSAR mode), or range resolution with average power, adaptively changing the chirp bandwidth and peak pulse power and/or pulse duration. The spacecraft design is anyway determined by the most demanding hi-resolution tasks.

C-band SAR trade-offs results

Parametric evaluations were also performed at C-band, achieving similar results in terms of swath widths, which are anyway superiorly limited by the antenna width.

Substantially lower RF average transmitted powers, as well as DC power requirements, are needed consistently with the lower range resolution feasible at C-band. A SCANSAR mode would equalize the azimuth and range resolutions while nearly doubling the width of the imaged ground strip w.r.t. a hi-resolution X-band implementation. In this way a simple medium resolution SAR could be implemented for wide area surveillance tasks.

4.2.4. SAR antenna requirements

A phased array with electronic scanning in the elevation plane will position the swath within the access angle. Besides providing beam steering in a +-15° w.r.t. the antenna normal, phase control must also provide beambroadening in the elevation plane to match antenna beamwidth to the swath position inside the access angle.

Due to the aperture oversizing factor of about 2.1:1 at minimum off-nadir angle, limited beamshaping in the elevation plane can be also implemented to equalize the antenna gain through the swath width. The higher beamslopes outside the swath will improve range ambiguity control, which is particularly important at low incidence angles. Besides, nadir echo suppression requires putting a beam null at nadir.

Achieving such features with phase control only (constant amplitude illumination being preferred to reduce antenna width at maximum off-nadir angle) may be an untrivial task, but it is feasible. At X band a passive single polarization design, using multiple panels of radiating waveguide slots fed by power dividers carrying embedded phase shifters, can be realized starting from already proven but simpler design [5], at a specific weight of 10 to 12 Kg/m^2 with less than 1.5 dB losses. A probably lighter technology can also be implemented at C-band.

5. DATA TRANSMISSION

Data transmission capabilities are very important to fulfill the observation missions herein considered. Two main approaches were considered: direct data transmission to ground and the use of data relay satellites. On board storage with subsequent data dump was not considered practical and effective, also due to expected near-term technology limitations.

5.1. Direct transmission to ground

The simplicity of this approach is partly offset by coverage limitations, which render direct transmission unsuitable when performing SAR observations over sites too far apart from the data station.

In a tactical scenario, however, opposing forces are normally deployed within a circle a few hundred miles wide. Data receiving stations deployed within, or close to, the theater can easily access the satellites of the constellation during overpasses and get real-time data for immediate, on-site, ground processing. Table 5.1.1. summarizes the projected characteristics of a direct trasmission system at X-band.

** Satellite terminal
- Frequency: 8 GHz
- Antenna: mechanical steering, driven by on-board navigation system;
- EIRP : 27 dBW (Ant. gain:[illegible] dB; Tx pwr: 10 W);
- Data rate: up to 200 Mbit/sec;
- Modul./coding: QPSK/ > 4 dB coding gain;
- Mass and DC power: 10 Kg, [illegible]0 W

** Receiving Station
- Antenna: transportable, with tracking;
- G/T: 16 dB/K° (Ant. diam.: 1.8 m);
- Slant range: up to 2400 Km;
- System margin: > 3 dB

Table 5.1.1. Direct Data transmission to ground: System performance

For a single plane sun-synchronous orbit constellation, meeting the design criteria discussed in 3.1.1.. The accessible area from the satellite passes in visibility of a data receiving station has been parametrically evaluated for a minimum elevation angle above the horizon of 5°. It must be noted that the above area can be accessed two times per day and per orbit plane: multiple orbital planes constellations will increase the dayly number of such 'events'. Table 5.1.2. shows the number of usefull satellite passes and total accessible area vs. orbit altitude. The total available access time is between 2500 to 3100 sec., per 'event', during which 500 to 600 images typically 30 by 30 km wide, could be made available to the data station. Such values may, obviously, be reduced due to real-time ground processing limitations.

Orbit altitude (Km)	Number of satellite passes	Accessible area (10^6 Km^2)	Total access time(s)
268	10	4.9	2500
362	9	7.6	to
460	7	9.8	3100

Table 5.1.2. Direct transmission to ground mission related performance

5.2. Data transmission via Data Relay Satellites

For strategic observations over sites not in direct visibility of ground stations, the use of a DRS network will be a cost-effective and performant solution. Our studies have shown that , in view of the severe lightsats volume and mass limitations, Inter-Orbit Links should preferably be carried out at 60 GHz. Efficient coding schemes providing > 4 dB coding gain must also be used to further decrease antenna size and power requirements on both user terminal and Data Relay Satellite. The DRS repeater will feature onboard demodulation-remodulation, while decoding is performed on ground.

Table 5.2.1. outlines the key characteristics of a DRS transmission system based on 60 GHz user-to-DRS links and the 20 GHz band for DRS-to-ground links.

** Satellite terminal:
- Frequency: 60 GHz band
- EIRP: 57.5 dBW (1.5" ant.diam.; 10 W Tx pwr)
- Mod./coding: QPSK/ >4 dB coding gain;
- Data rate: up to 200 Mbit/sec.;
- Mass and DC power: 15 Kg, 70 W

** DRS payload:
- Frequencies: 60 GHz Rx; 20 GHz Tx;
- G/T: 33 dB (2.1 m ant.diam. at 60 GHz);
- EIRP: 56 dBW (1 m Tx ant.);

** Data Station:
- G/T : 27 dB/K° (12" ant.diam.);
- Slant range: to synchronous orbit;
- System margin: > 8 dB

Table 5.2.1. Data transfer to ground via Data Relay Satellites

6. PLATFORM DESIGN REQUIREMENTS

To complete this overview on lightsat constellations, the satellites control aspects and the main platform design considerations are addressed in the following.

6.1 Orbit control

Accurate orbit control is important for two reasons:

- to take images of specific targets, when the satellite is not directly visible from ground, according to preloaded time-related instructions. Concerning this point a positional error contribution of up to 0.3 to 0.5 Km could be tolerated, which will decrease the effective imaged area by less than 5 percent. Accurate orbit keeping, in presence of drag and other external forces, is mandatory to avoid coverage gaps due to differential lags between satellites of the constellation, and may be implemented with electrical propulsion on long duration missions. Orbit keeping in a tight altitude deadband requires thrust modulation which will benefit from the presence of an autonomous navigation system. However long lifetime operation of electric thrusters has to be demonstrated.

-To support removal of satellite orbital position related errors during ground image processing. For high resolution X-band SAR imaging, the tolerable error in the orbit position estimate is in the range of 300 m, above which SAR imaging degradations will occur.

Implementing a GPS-based autonomous navigation syste may achieve a real-time three dimensional orbital position restitution with 100 to 200 m accuracy, thus meeting the above requirements.

6.2 Attitude control

Spacecraft attitude control impacts two performance:

- the accuracy with which the swath can be positioned inside the access angle;

- the quality of SAR images after ground processing.

It is important to distinguish between attitude control and attitude restitution. With SAR electronic beam repointing, satellite attitude errors in roll affect mainly the swath positioning and can be partly corrected if the roll attitude is known with a better accuracy than the spacecraft attutude control. Yaw and pitch attitude errors impact the image quality and must be strictly controlled. If their restitution is known with an accuracy 3 to 5 times better than their control, error removal during ground processing will be easier. Under these assumptions, a 0.1° attitude error in roll control and less than 0.08° in pitch and yaw control can be set as reasonable goals.

Nevertheless high frequency attitude error components, as might be induced by flexible appendages subjected to transient forces, must be minimized due to the bad effect they may have on image quality.

6.3 Spacecraft platform requirements

We outline some platform subsystems design aspects most relevant to SAR performance and operation.

Low orbit altitudes will require electric propulsion, which must be considered an enabling technology [3] for these SAR lightsats missions, as already discussed. Nevertheless, flying satellites at low altitude has also positive effects, in particular concerning the more benign environment [5].

The structure design, departing from conventional, should adopt a highly integrated approach where a truss frame utilizes important SAR payload components, like the antenna or big boxes, as part of the structure itself, in order to save mass. Since the bulk of the heat is produced by the SAR HPA, heat rejection should be by direct radiation to space as far as practical: this is certainly easier on down-dusk orbits.

The solar array design is strongly dependent upon the SAR payload operating duty. Infact, since the SAR will normally operate for short intervals totalling 5 to 10 % of the orbit period, payload supply can normally be from battery, the solar array serving mainly for battery recharging. The orbit plane choice, and the percentage of time spent in Earth-shadow per orbit period, will also impact the solar array sizing to provide the required energy. Besides, electric propulsion DC power requirements, will also increase the solar array sizing. Accordingly, the solar array type may range from a fixed wings configuration, suitable for satellites in sunsynchronous down-dusk orbits, to a sun-tracking configuration for spacecraft in low to medium inclination orbits.

On the above grounds, long lifetime operation of mass-efficient batteries, subjected to rather deep and periodic (up to 30000) discharge-recharge cycles is an oustanding issue for such L.E.O. observation lightsats equipped with SAR sensors.

Spacecraft telecommand must be secure, and possibly jam-proof, to avoid unauthorized entries to the satellite system. Data encryption must be also implemented, in spacecraft telemetry and SAR data transmission to ground, to prevent eavesdropping by unauthorized users.

7. CONCLUSIONS

Lightsats in the 500 to 800 Kg range can carry SAR sensors for high resolution (2 to 3 m), short revisit interval, tactical observation missions. Medium resolution (order of 5 m) missions, for strategic and, in general, international law-enforcement applications, are also feasible. Such spacecraft can form permanent constellations of, typically, 6 to 8 satellites per orbit plane to provide global coverage. Multiple orbit planes constellations can offer enhanced performance, allowing also a gradual system build-up. Smaller constellations of 4 to 6 spacecraft can be also implemented to cover a narrower latitude belt around the equator, while significantly improving the average revisit intervals at sites close to the orbits inclinations.

In summary, SAR lightsats constellations can offer certain performance unmatched by existing, or planned, single and larger observation satellites and can provide a valuable answer to specific operational military needs in both tactical and strategic scenarios. The required technologies exist, with few exceptions needing an early in-orbit demonstration of devices being now developed. Nevertheless, advances in few fields will certainly benefit future generation lightsats.

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A TACSAT SAR Concept

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Abstract

The payload concept covered by this paper is that of a low cost, high performance radar sensor capable of detecting and recognising static objects within an imaged scene of the Earth's surface using the Synthetic Aperture Radar (SAR) technique . The overall system is integrated with a TACSAT platform in Low Earth Orbit (LEO) and, although only passing reference is made to this feature in the paper, the radar could also have a capability for the detection of Ground Moving Targets (GMTI).

The paper provides a parametric review of such a sensor in the light of the target and background features to be observed. A design concept is included showing the possible hardware realisation of a candidate system, as well as budgets for the mass, size, power and pointing requirements of the instrument. Additional design features considered are the influence that short duration missions may have on hardware redundancy and the effect of short, low duty-cycle observation periods on solar array and battery sizing.

The paper concludes by pointing the way towards a low cost R and D demonstrator system allowing a practical investigation of the key techniques and technologies.

1 Introduction

As a result of the recent UK ASTOR programme and the deployment of US systems such as JSTARS and ASARS, the ability to combine information derived from high resolution imaging radars together with the detection of ground moving targets has been demonstrated to afford an extremely valuable Military capability. As a consequence there is continued development of these systems together with refinements in their operational deployment and usage. However, a number of restrictions exist in the deployment of these systems associated with their maximum operating range. The maximum operating range is governed by the height at which the platform can fly, so for example a radar on an aircraft with a ceiling height of 30,000 feet will only be able to operate out to about 250 Km before grazing incidence is reached. This limitation can be removed by placing the sensor in a space-based platform. In order to maximise the military utility of the information produced by the instrument and to allow recognition of targets, a design goal of 1m has been assumed for spatial resolution.

The potential advantages of extensive, unrestricted fields of view afforded by space based sensors makes then extremely attractive for wide area Military surveillance and hence worthy of further attention, particularly with a view to replacing airborne systems (although it should be recognised that there are a number of inherent technical difficulties in designing a spaceborne system that has similar sensor capabilities).

Large areas of the Earth's surface can be observed on a periodic basis and more detailed information may be obtained by dwelling on selected areas. Radar also has the added advantage of all weather day/night operation. Recently imaging radars (SAR) have begun to demonstrate the capability achievable from space-based systems but use relatively heavy platforms (eg ERS-1 weighs 1275 Kg although it contains 4 other sensors as well as the SAR) and hence require large and costly launch systems and infra-structures. Further they have relatively poor resolutions (eg ERS-1 is approximately 30m) making them unsuitable for most Military applications and have no inherent capability for the detection of ground moving targets.

It is, however, possible to conceive SAR sensors for integration with various forms of small satellite platform and depending on the type of platform used to carry the sensor, different options exist for its launch. With one option, the satellite could be launched under the control of a Military commander from a mobile, low-cost launch vehicle. With another, the satellite can be launched by a major launch vehicle. for instance in conjunction with the launch of a major payload making use of residual mass capability. These options potentially would allow the sensor to perform both a short term tactical role, including the replacement of damaged or destroyed satellites in times of conflict, and a longer term strategic role whose tasks might also include training of users of these highly flexible systems.

The small size and low mass of the satellite and sensor systems considered in this paper will result in significant cost savings making the deployment of constellations of satellites very much more attractive. This has the further advantage of reducing revisit times and consequently providing information not otherwise available from any other source, and of improving system survivability. In the next section some of the options for a sensor concept are described. This is followed by a consideration of system aspects and cost and schedule drivers.

2 Impacts of a Potential TACSAT SAR Mission on Satellite Design

2.1 A Potential TACSAT SAR Mission

The TACSAT mission is one which is associated with a Theatre conflict where the Theatre is defined as being a region measuring some 2000km by 2000km and the duration of the conflict is anticipated as being relatively short, typically of months, rather than weeks or years. Although areas of political and military risk can lie at any latitude, the ending of the Cold War suggests that if there were to be such a Theatre conflict, then its occurrence in the lower latitude regions, rather than the more Northerly latitudes, seems more likely.

The mission envisaged for a TACSAT SAR is to provide the Theatre Commander with a dedicated service yielding surveillance data only over the crisis area. In this paper, the dedicated nature of the TACSAT mission has been taken to mean that the Theatre Commander has total control over the satellite so that the satellite will not be made available to provide surveillance over any region other than this Crisis area. Depending on the particular operational philosophy, satellite control might be exercised either directly from within the Theatre, or remotely using global control systems but under the primary control of the Theatre Commander.

Finally, the dedicated nature of the service proposed for the TACSAT SAR facilitates some interesting departures from the system thinking associated with global space surveillance concepts: these departures influence the choice of orbit, the spacecraft bus design, and the basic sensor design.

2.2 Orbit Parameters

The expectation of a Theatre location away from the Northern latitudes invites consideration of orbits of a smaller inclination than the polar sun-synchronous orbit which is characteristic of civil remote sensing missions. In particular, if the chosen operating philosophy for the TACSAT SAR is one of launch-on-demand in time of crisis, then the orbit inclination can be chosen to maximise viewing opportunities over the crisis area. If, however, the philosophy is one of in-orbit storage then an earlier decision must be made about inclination because of the large fuel overhead involved in effecting changes in orbit inclination. However, even in this latter case, it may well be appropriate to set the inclination of the storage orbit so that coverage is optimised for the smaller latitude regions. The absence of sun-synchronism, consequent on the small orbit inclination, will not present a particular problem to the radar sensor which is capable of operating as well by night as by day.

In addition, the relatively short duration envisaged for a Theatre conflict allows serious consideration to be given to orbits significantly lower than those used for long duration remote sensing missions: these lower orbits, perhaps as low as 250km, would be suited to a philosophy of launch-on-demand. However, although particularly low orbits are less attractive for the case of in-orbit storage because of the short life time which results from atmospheric drag, it is possible to conceive two distinct and attractive patterns of use for a TACSAT using the philosophy of in-orbit storage.

During the storage period, the TACSAT can be used for training purposes as well as for active surveillance service although, at coarser sensitivity. In the event of crisis, the potential would exist to reduce the orbit altitude in order to improve sensitivity over the operational Theatre. It is important to note however, that such a reduction in altitude will not bring significant advantages in improved spatial resolution, and the coverage access available to the satellite at any one pass over the Theatre will be more limited than that achievable from a higher orbit.

2.3 Duration of Service

The dedicated nature of the surveillance required from the satellite means that the periods during which service is required are of short duration. Typically, a satellite in the Low Earth Orbit (LEO) necessary for SAR operations takes only 5 minutes to traverse a Theatre measuring 2000km on the side and with only 2 such passes likely per day, payload power demands can readily be met from batteries. The advantage of this to the platform design is that relatively small solar arrays can then be used to collect electrical power and trickle-charge the batteries during the prolonged periods of payload inactivity.

2.4 Sensor Design Goals

The design goals for the sensor concept are shown in Table 1. A major difficulty in realising a radar design of this sophistication is the achievement of such high resolutions and in the implementation of a phased array antenna able to operate over such wide instantaneous bandwidths.

Synthetic Aperture Radar (SAR) is a sideways looking coherent imaging technique capable of producing map-like imagery. SAR employs different principles for achieving resolution in the two dimensions of the imagery. Conventional high resolution radar techniques such as transmitting a wideband, long duration pulse and correctly matching to it on receive are employed in the dimension perpendicular to the satellite track. Table 2 illustrates the bandwidths commensurate with resolutions for a variety of grazing angles.

The limitations on achievable bandwidth are due to the capabilities of waveform generators and digitisers and the dispersion through the transmit and receive chains. Current technology imposes a realistic limit of about 1 GHz. Unless changes can be agreed, a further limiting factor will be the current operating band allocations defined by the World Advisory Radio Council (WARC) which impose restrictions of 300 MHz at X-band (9.6GHz) and 600 MHz at J-band (13.7GHz). Dispersion in the atmosphere and ionosphere is not thought to present a major problem and, if present, the effects could be removed using techniques equivalent to autofocus. Overall it is felt that a resolution of 1m at X band is feasible at a 30° grazing angle.

The synthetic aperture technique is employed in the dimension parallel to the satellite track. This involves the coherent addition of many pulses to synthesize an aperture sufficiently long to provide the desired resolution on the ground. This synthesis is only possible if the variation in phase path from the sensor to the target and back is known to a fraction of the radar wavelength along the synthetic aperture. This implies that very accurate knowledge of the orbital path of the satellite is required. However, for resolutions of the order of 1m no measurement method of the orbit gives sufficient accuracy in realistic timescales. To surmount this problem it is expected that autofocus techniques will have to be employed. These have the added benefit that they will also automatically correct for other effects which can alter the phase path length, such as atmospheric, ionospheric and gravity perturbations.

3 TACSAT SAR Satellite Concept

3.1 Launch Options

A wide range of current and planned vehicles is available to members of NATO, optimised for the launch of satellites with masses as small as 100kg and ranging upwards to many tonnes. Vehicles such as Pegasus and Scout can lift the lower end of the TACSAT mass range, around 250kg, into a 500km orbit.

However, there is a significant gap in mass-optimised launch capability such that the next available vehicles are San Marco Scout and Taurus which, when developed, will be optimised respectively, for the launch of satellites of 750 kg and 1200kg into 500km orbit. This gap in capability of current launchers means that, while there is not a technical problem in launching into LEO, satellites whose mass lies at the upper end of the TACSAT range, there could be an appreciable cost penalty if Taurus were used. Whereas launches on Pegasus and Scout are priced at $10M and $18M respectively, the cost of launching on Taurus is predicted to be $30M.

An alternative and cheaper approach to the heavy TACSAT launch issue is to make use of the "residual mass" capability often associated with the launch of major satellites on large launch vehicles. Residual mass is the mass difference between that of major satellite and the maximum launch mass capability of the proposed launcher. It is anticipated that satelites will be launched in this way for a cost of $20k - $30k per kilogram, depending on launch configuration

It is felt that availability and cost are likely to have a major impact on the viability of TACSAT concepts so that for the heavier TACSATs, the advantage of a residual mass launch would be considerable. However, the penalty that associates with this launch philosophy is that launch-on-demand is not available. Therefore with this concept, the possibility of designing for short mission lifetimes is ruled out because the satellite must be stored in space perhaps in a higher long life time orbit so that it can be brought down to operational altitude as and when an appropriate crisis develops.

As a result of this understanding of the realities which we perceive as constraints on the mission, the SAR concept described in this paper is based on use of the residual mass launch option.

3.2 Platform Options

The accommodation volume available for a residual mass launch is basically toroidal and therefore the bus structure proposed as a platform for the SAR is of toroidal form. An additional benefit which accrues from this bus structure is the potential for launching several satellites simultaneously from the same launch vehicle.

The toroidal structure is therefore capable of providing both an economical demonstration of the technology through a residual-mass launch opportunity, and the potential for launch of a full constellation via a dedicated launch - without the need for any satellite redesign between demonstrator flight and full constellation.

The outline concept of a SAR mounted on such a toroidal platform is presented in Figure 1. Budgets showing the mass and power demands likely to be associated with such a satellite are presented in Table 3.

3.3 SAR Options

Three distinct options of SAR design can be envisaged to provide a tactical capability. In order of rising complexity, these options include, mechanically steered systems using a single central high power amplifier (HPA), electronically steered systems like RADARSAT which use a single central HPA and ferrite phase shifters for beam deflection, and electronically steered systems like the European ASAR which use fully active phased array technology to provide Scansar and Spotlight operating modes.

Of these fundamental design options, two were considered as candidates for discussion in this paper, a partial phased array providing electronic steering only in the azimuth direction and a fully active phased array providing both azimuth and elevation steering.

The case of the potentially simpler, partial phased array where the elevation steering is provided only by mechanical pointing was considered, but it was concluded that because the time taken to mechanically repoint the system would be on the order of minutes, the number of useful Spotlight imaging opportunities available to the Theatre Commander would be limited to a rather small number. When this number is compared with the much larger number of such opportunities potentially made available from a fully active phased array, there appeared to be a strong case in favour of the fully active system.

Thus, although neither of these cases has been analysed quantitatively, it is felt that the flexibility of coverage and operating modes which can be derived from active phased array technology is of such benefit that its use, even in a small tactical satellite, verges on the mandatory. Therefore, the bulk of this paper concentrates upon this type of sensor but acknowledges that designs based on passive antenna technology of the ERS-1 type may be able to provide a more limited capability but at a smaller mass.

3.4 Platform Orientation

It is envisaged that during the long periods of inactivity between surveillance tasks, the platform orientation will be controlled in order to provide a benign thermal environment, and that the whole platform will be rotated when necessary in order to provide the appropriate viewing angle, and then returned to the benign orientation after the short period of active service.

The technique envisaged to effect the necessary rotation manoeuvres from benign to service positions is one of use of the large torques which can be produced from control moment gyros. In this technique, a caged system of at least three gyros is set up and the angular momenta of each is established so that the net angular momentum seen outside the cage is zero. However, if a torque imposed from outside the gyro cage tries to repoint the direction of any of the component angular momenta (the direction of a gyro rotation axis), then that torque has to be considerable and can be used to produce a strong turning moment on the spacecraft.

A difficulty with this philosophy is that the vehicle is exposed to a significant thermal disturbance during the manoeuvre as a result of solar illumination. However, the manoeuvre can be designed to be of a sufficiently short duration that the total temperature excursion may be limited to acceptable bounds.

The advantage of this approach to satellite rotation movements is that although it calls for electrical energy is required, such energy can be supplied indefinitely (via the on-board batteries) from the solar array. For this mission, where many roll manoeuvres are envisaged and speed is likely to be important, the approach compares favourably with the use of cold gas thrusters which provide a finite life to the total mission.

The advantage of this operational philosophy is that the manoeuvre enables the SAR to view regions on either side of the nadir track. Such an operational philosophy requires that the whole satellite including platform, solar arrays, and SAR antenna, be designed to form a robust structure with no flimsy protrusions.

4 TACSAT SAR Sensor Concept and Performance

The paper has investigated the impact of different operating modes and the fundamental SAR parameters on the performance achievable for a TACSAT SAR mission. These parameters include, operating altitude, RF carrier frequency, spatial resolution, and possible antenna structures. However, in order to limit the scope of the paper, all options have been based on an assumed antenna area of $6m^2$. This area is derived from considerations of viable payloads for implementation onto a toroidal platform within the TACSAT mass goal of < 700kg.

The detailed performance of a SAR is described in this section, specifically for the single-look capabilities which will characterise its performance in Spotlight mode, by parameters which include;

* sensitivity expressed in terms of noise equivalent σ_0 (NEσ_0) and presented for the worst cases situation which is associated with the furthest edge of the available coverage width

* access coverage width as that distance from nadir beyond which ambiguities rise to an unacceptable level

* available swath width for various operating modes

* spatial resolution in azimuth and elevation for various operating modes

The operating geometry for a TACSAT SAR is shown in Figure 2 in order to identify the principal parameters.

4.1 Operating Modes

Spotlight

In order to perform a useful reconnaissance role, the SAR must be capable of providing images of specific target zones such as airfields, harbours, choke points and railway installations at the finest possible resolution. This task calls for Spotlight operation in order to provide adequate resolution in the along-track (azimuth) direction and the provision of this capability is a major driver in the design of the antenna.

In principle, the azimuth scanning associated with Spotlight operation can be provided by mechanical steering of the antenna. However the difficulties associated with repeating such mechanical steering several times during a single pass over the theatre are believed to be so great that electronic steering in the azimuth direction is considered to be essential.

Steered SAR

In addition, it is envisaged that a TACSAT SAR may also be required to provide surveillance over a larger region. Such cover can readily be provided in the form of a continuous strip-map using the conventional Steered SAR mode which has been characterised by SEASAT, ERS-1 and the SIR missions.

Scansar

If even wider area coverage is desired, then the possibility exists of using the Scansar mode which will be demonstrated on the RADASAT and ASAR missions. However, provision of a Scansar capability relies on the ability to rapidly redirect the antenna beam in the elevation direction: such a capability requires the provision of electronic steering in the elevation direction.

Ground Moving Target Indication (GMTI)

In this mode, which may be combined with Scansar, interferometric techniques are envisaged as providing detection of surface targets in motion. The data will be processed on board so that only the detections need be down linked back to the ground data receiving station. However, it is anticipated that problems with revisiting may lead to difficulties in interpretation and thus requires further research.

4.2 Operating Altitude and RF carrier Frequency

Access coverage width is fundamentally limited by system parameters which include; antenna area, slant range to ground, RF carrier frequency, and satellite velocity - although there is no significant impact from satellite velocity which changes but little with altitude. For this case in which antenna area has been constrained to $6m^2$, the dominant parameters are slant range and RF carrier frequency.

The effect of changes in altitude on access and sensitivity is shown in Figures 3 and 4. The data are presented for the case of operation in Spotlight mode with an assumed bandwidth of 300MHz. Although this bandwidth is characteristic of the WARC limit at X-band and is in excess of the allocations at S and C, it is nontheless interesting to use a single bandwidth in order to see the fundamental parametric trends. It is also important to note that although a mean RF power of 750W is needed to ensure adequate sensitivity throughout most of the viewing range, such high mean powers can be provided during Spotlight operations by increasing the duty cycle well beyond the few percent normally used for wider swath observations. In this way, peak powers in the T/R modules can be limited to less than 4W.

It can be seen that at the lower frequencies, increasing altitude provides relatively small improvements in access capability while at the higher frequencies, substantial improvements can be realised. However, in both cases, increasing altitude significantly degrades sensitivity. The sensitivity data presented in Figure 4 show the worst case sensitivity; i.e. at the furthest distance from nadir for which acceptable ambiguity performance can be achieved. Data showing the variation of sensitivity achieved from an altitude of 500km for both C and X band is presented in Figure 5.

A further impact of increasing altitude can be seen in Figure 6 where the impact on the across track component of spatial resolution, range resolution, has been presented as a function of distance from nadir and operating altitude for a single RF chirp bandwidth - 300MHz.

Selection of Operating Frequency

In selecting an RF carrier frequency for a space-borne high resolution military SAR, several factors must be taken into account. First, it must be noted that, in principle, any ground range resolution can be achieved by provision of a radar emission pulse of adequate bandwidth - typical examples are shown in Figure 6 for the case of 300MHz bandwidth allocation, but international agreements constrain the bandwidth allocated to space radars as indicated in Table 2b. However, although such agreements might be breached in times of crisis, the TACSAT mission concept also includes peacetime use and for this, adherence to bandwidth allocations is essential.

The figures presented in Tables 2a and 2b indicate that only an X or J band system would be capable of providing a ground resolution close to 1m, whilst the use of L, C or Ka band would fail to provide a ground resolution better than 3m over most of the grazing angles of interest.

In addition to the international regulations, other factors affecting the choice of RF carrier frequency are as follows;

- the need for the SAR wavelength to be significantly less than the image resolution. This is likely to exclude the use of L-band (0.23m wavelength) for a system aiming to achieve around 1m resolution.

- ionospheric dispersio·. This is a frequency dependent phenomenon which can cause a significant degradation in image quality for frequencies around or below L-band.

- atmospheric dispersion. This is generally, not a problem for frequencies below X-band, but may degrade image quality at higher frequencies, e.g. around the water vapour resonance frequency of 22GHz.

- atmospheric attenuation. This varies with frequency, becoming more acute for frequencies above X-band, possibly ruling out J-band and probably ruling out Ka-band. However, the attraction of the very wide bandwidth allocation at J-band and the associated potential for very fine spatial resolution may mitigate in its favour.

- poorer efficiency of solid state power amplifiers at higher operating frequencies These factors indicate that X-band is likely to be an optimum frequency for very high resolution (around 1m)military surveillance SAR, with C or possibly S band representing a viable alternative providing somewhat poorer resolution.

4.3 **Imaging Capabilities**

For the TACSAT SAR mission considered in this paper, target imaging requirements may be considered to fall into the following two main categories;

(i) detection and classification of hard targets, e.g. military vehicles, tanks, grounded aircraft, ships, etc.

(ii) monitoring and change detection for distributed targets, e.g. airfields, military camps and installations, harbours, railway yards, choke points, etc.

In the former cases, the targets of interest generally have large radar cross sections relative to the background scene. As a result, the ability to detect and subsequently classify such targets depends primarily on the spatial resolution of the imagery, the system radiometric performance being relatively unimportant. A spatial resolution of at least 3m, and ideally less than 1m, is considered necessary.

In the latter case, however, the features of interest are generally no brighter than other features in the scene; the key requirement for discrimination and recognition of such targets is then, good radiometric resolution(best SAR speckle reduction). This may be achieved by multi-looking. The spatial resolution requirement for these types of target is considered less exacting than for "hard" targets at around 5-6m. However, the fact that multi-looking is required implies that the inherent single-look capability be similar to that for hard target detection (i.e. around 1-3m).

The SAR must possess a radiometric resolution adequate to allow discrimination between adjacent distributed features. A sensitivity characterised by a noise equivalent sigma zero (NEσ_0) of -23dB is considered necessary.

4.4 **Antenna Structures**

It can be seen from the preceding discussion, that the TACSAT antenna requires an electronic steering in azimuth in order to provide the Spotlight mode and that a similar capability in the elevation direction is highly desirable in order to facilitate the imaging during a single pass, of different zones located at different elevation bearings relative to the satellite.

Thus, the case for electronic steering in both azimuth and elevation directions appears to be strong and has led this paper to propose a fully active phased array solution for the envisaged TACSAT mission.

The particular aspect ratio assumed for the antenna is based on the need to accommodate around the toroidal bus, the 6m^2 overall area required to provide the unambiguous access demonstrated in the earlier discussion. Two distinct cases have been examined one at C-band, the other at X-band. In both cases, the antenna is built from 11 panels which are assembled onto the bus as indicated in Figure 1. In the C-band case, each panel is populated with 32 T/R modules and measures some 1300x418mm^2 whereas in the X-band case, each panel is populated with 64 T/R modules and measures some 1400x388mm^2. Thus, the aspect ratio of the antenna is similar in both cases being 4.6m(az) by 1.3m(el) in the C-band case, and 4.27m(az) by 1.4m(el) in the X-band case. It is important to note that despite their intended major role as Spotlight imaging sensors, both of these SARs would be capable of providing better than 3m resolution in conventional Steered SAR mode.

A feature of the active antenna approach which can be exploited in this design is the impact on T/R module design which results from the adoption of Spotlight as the principal operating mode.

When operating in this mode there is a need for the emission of a particularly high level of mean RF power in order to restore the link budget which is adversely affected by the increased noise level which results from the wide bandwidth needed to achieve fine range resolution.

Because the region being imaged is only a small proportion of the maximum, range-unambiguous swath, it is possible to increase the transmitter duty cycle and provide a several-fold increase in mean RF power output during Spotlight observations. Typically, increases of a factor of 5 can be achieved over that normally associated with civil remote sensing missions where the duty cycle is limited to around 5%. The advantage of this approach is that these increases can be achieved without resorting to an increase of the peak RF power handling capability of the T/R module output stages.

It is interesting to compare this TACSAT mission with a civil remote sensing mission. In a typical C-band civil mission, 320 T/R modules operate at up to 8 W peak RF power in order to provide < 150 W mean RF power, whereas in the Spotlight TACSAT mission considered here, 352 T/R modules operating at 6 W peak power can are operated to provide up to 750 W mean RF power. In the X-band case considered in this paper, 704 T/R modules are operated at less than 3 W peak power in order to provide the same mean RF power (750 W).

However, the technique is only applicable to achieving short term gains because the increased transmitter power requires that the system be capable of handling the associated increase in waste power. In this concept the thermal problem is regarded as a thermal impulse and handled by providing a sufficiently long gap between observations that the long term mean power handling capacity is not exceeded.

4.5 DC Power Services

It is envisaged that the TACSAT SAR will operate from batteries during periods of service and that during sunlight spacecraft operation, the batteries will be charged from the solar arrays at a rate sufficient to maintain them charged, ready for the next service period. Although the power demand during periods of service is very large, the total energy demand is modest and can potentially be satisfied from a low capacity battery, charged from the relatively small solar array.

It is estimated that the surveillance task will place a daily energy demand from the batteries amounting to only about 500watt-hrs. However, the battery must be sized not so much to provide the total energy load but rather, to satisfy the 6kW peak power/current demand which is needed to energise the SAR and the data down-link during the 5 second periods associated with each Spotlight observation. The demand for low internal resistance which results from the large peak power demand, favours the use of NiCd batteries rather than NiH_2. The mass associated with these NiCd batteries will be on the order of 40kg and this figure has been included in the mass budget shown in Table 3.

The total energy demand can be satisfied from a relatively small solar array whose area is not influenced by the large peak power demand. It is envisaged that this energy will be collected during the period of around 22 hours when the SAR is not in a position to observe the Theatre and that during this time a mean power collection rate of only 25 watts averaged over each orbit, will be sufficient to provide the SAR/down-link energy budget.

However, in order to provide power for the SAR and the additional 200W power estimated as being necessary for the satellite bus house keeping activities, and to account for eclipse periods in the charging cycle, it is estimated that a peak collecting capability of some 300 watts should be allowed at this stage in concept development. Such a power collection capability can be provided at end of life (notionally, 5 years) by $2m^2$ of solar array, assuming the use of GaAs solar cells. Moreover, since the operational concept assumes that during this period, the whole spacecraft is placed in a thermally benign orientation, this orientation can be chosen so that the solar array area can be optimised for smallest mass by arranging for it to point continuously at the sun.

4.6 Data Down Link

<u>Data Rate</u>

Different data rates can be associated with the SAR depending on whether range compression is conducted on-board or on the ground. With the long duty cycles envisaged for use in Spotlight mode, there is a need to receive radar returns, not only for the duration of the time delay from near to far sides of the image region, but also for the duration of the transmit pulse.

The classical technique for reducing SAR down link rates is to accept all the returns from a given single pulse emission into a register (during a receive window period which is much longer than the transmit pulse period), and then to transmit the data accumulated in that register during the longer period associated with one pulse repetition interval. However, when the receive window is shorter than the transmit pulse, there is less opportunity for data rate compression. If, on the other hand, pulse compression is conducted on-board, then the duration of the receive window - post range compression - is identical with the time delay between near and far sides of the image region. This time difference can be significantly smaller than the interpulse period and appreciable reductions made to the data rate.

Using such techniques, it is possible to limit the data rate to values as small as 130 Mbit/s for operations at C-band where the SAR chirp bandwidth is limited to 100MHz, and to 400Mbit/s for operations at X-band where the chirp bandwidth is limited to 300MHz. Data links with these bandwidths exist but the problems of handling these quantities and rates of data will require further examination.

Ground Segment

The operational concept for the TACSAT SAR is that it should make its observations and almost simultaneously transmit the SAR data to ground for reception within the Theatre command centre where the data will be processed into an intelligible image.

It is envisaged that the ground data reception station will operate at elevation angles greater than 5° using a 5m diameter, steered dish, will operate at Ku-band with a G/T ratio of 29dB/K. Indeed, the data could even be routed into existing ground exploitation stations developed for airborne reconnaissance systems thereby effecting a significant cost saving.

Ground Processing

Currently available ground processing algorithms will process TACSAT data in its Steered Beam and Scansar modes. Further development could be required for the high resolution Spotlight mode to cope with the rapidly varying Doppler aliasing at 1m resolution, especially at X-band, and autofocussing will be required to maintain focus over varying terrain height. Existing airborne techniques are not applicable.

Data is received at 10^8 samples per second: Spotlight processing with autofocus will require some 10^3 FLOP per resolution cell, and therefore, real time processing could require a computer power of over 100GFLOPS. Currently, this would require an array of some 5000 vector processors and could be a state of the art machine, posing major data handling problems. However, assuming a deployment date at least 5 years in the future, improvements in processor speed of at least 4 times can be anticipated.

If 40 minutes, rather than real time, is allowed to process the maximum data anticipated from a single pass, then the number of vector processoring nodes required can be reduced to 125. Such a processor could be housed in a single 19inch equipment rack and would be equivalent to top-of-the-range commercial equipment.

Space Segment

The impact of this type of ground network on the space segment is that the down-link transmitter must be capable of providing an EIRP of 16dBW from a small unsteered antenna. Such an EIRP can be provided from a 120W TWTA assembly whose total DC power demand is likely to be around 300W including power to the TWT power conditioning electronics and the RF signal drive electronics. Mass of the downlink equipment when supplied in dual redundancy is estimated to be around 17kg including the shaped beam antenna.

5 Programme Cost/Schedule Drivers

5.1 Launch Costs / Constellation Size

There is much discussion about the possible lifetime of a TACSAT mission. Depending on the mission concept, the intended lifetimes can vary from several weeks - associated with launch on demand, through months - associated with launch in anticipation of a crisis, to years - associated with pre-deployment of the satellite(s) and storage in space.

The case in favour of launches using the residual mass capability often associated with large launch vehicles has been made in section 3 of this paper. Here, we discuss the possible impact of that launch philosophy on reliability required of a TACSAT SAR.

The particular needs which might be envisaged for a TACSAT SAR are that it be capable of viewing the designated Theatre at very regular intervals. For instance, a full system may be required to provide viewing opportunities at 8-hourly intervals: such a revisit rate would call for the service of a constellation of 3 satellites.

The toroidal bus design discussed earlier facilitates the launch of a mini-constellation of 4 or 5 satellites from a single, dedicated launch using Ariane-4 or Titan. System reliability can in this case, be improved by the 3 from 4 (or 5) redundancy which results from using the multiple launch.

5.2 Mission Duration and Component Reliability

The use of Flight qualified components coupled with exhaustive ground testing at every stage of the satellite development programme brings benefits in the form of minimised risk of mission failure, and enhanced reliability throughout the mission. However, such build procedures place a considerable premium on programme cost.

In recent years, industry has made progress in the streamlining of on-ground testing in order to optimise mission reliability versus programme cost and schedule. However, industry rigorously retains the service of flight qualified components for all programmes.

It is interesting to recall that flight components are in general drawn from the same production lines which deliver Mil Std components. The premium charge made for flight components results from the additional tests and rigorous traceability which is imposed on the basic Mil Std components to provide the greater level of confidence demanded for their use in space.

Two particular features of typical space programmes favour the high level of caution implicit in the dedicated use of fully qualified components. One is the long life usually demanded of the system; the other is the one-off nature of most (large) satellite systems. The first of these features demands the greater level of confidence associated with flight qualified components in order to guarantee reliability over the (long) life time; the other demands high intrinsic reliability because with a one-off system, there can be no resort to the N from M option for reducing the overall risk of failure.

Notwithstanding these cautions, in the case of launch on demand, the life time expected of the satellite can be quite short - typically around 6 months - and the potential advantages of using less highly qualified components is worth consideration.

However, when the proposed launch philosophy calls for the use of large launch vehicles, the long lead times usually associated with launches on large vehicles means that possibility of a rapid turn round from launch request to launch event is small. Therefore, the satellite must be designed on the expectation of a relatively long life and in this case the use of less highly qualified components is less desirable.

6 Conclusions

The capabilities of a TACSAT SAR have been reviewed in the context of notional Theatre surveillance mission. Of various options, it has been argued that only a SAR based on active phased array technology will provide the flexibility of performance needed to satisfy the notional mission.

It has been shown that a SAR operating at X-band would be capable of providing single-look imagery at 1m spatial resolution and a noise floor of better than -20dB when flown in a circular orbit at 500km altitude. The total mass of a satellite carrying this SAR has been estimated at around 600 to 700kg.
While it is quite within the bounds of technology to conceive integration of either of the SARs discussed in the paper on a platform suited to individual launch, it is specifically the absence of dedicated launch vehicles optimised for single launch of such a satellite mass which has been taken as a driver towards a toroidal bus structure suited to a multiple launch making use of the residual mass often associated with launches on larger vehicles.

The adoption of such a philosophy has been shown to provide the opportunity of getting single satellites into orbit economically but not with a rapid turn-round. In addition, the toroidal bus structure provides the possibility of deployment of a complete constellation, using a single launch from a large vehicle. In this way it is possible to amortise the cost of the larger launch against the corresponding cost of several individual launches using smaller vehicles.

In order to fully understand the operation of, and interplay between, the various components which comprise a TACSAT system and to optimise the design of a military constellation, a demonstrator system is clearly required. In particular, the use of active phased arrays in space could be demonstrated and overall system performance evaluated. This paper has shown that such a demonstrator is perfectly feasible using today's technology.

7.0 STATEMENT OF RESPONSIBILTY

Any views expressed are those of the author(s) and do not necessarily represent those of HM Government.

RADAR SPECIFICATION

PARAMETER	VALUE	UNITS	COMMENTS
operating frequency	9.6	(GHz)	X-band
spatial resolution	10 1 20-50	(m) (m) (m)	Steered Spotlight GMTI
minimum detectable RCS	10	(m^2)	
minimum detectable velocity	6.5	(knots)	radial velocity
P_{fa}	10^{-6}	(-)	
Polarization	(-)	(-)	scattering matrix preferred

Table 1 Design Goals for TACSAT SAR

RESOLUTION (m)	CHIRP BANDWIDTH (MHz) Grazing Angle 30°	45°	60°
10	26	32	45
3	87	110	150
1	260	320	450
0.3	870	1100	1500
0.1	2600	3200	4500

Table 2a Spatial Resolution Vs Chirp Bandwidth and Grazing Angle

BAND NAME	FREQUENCY RANGE (GHz)	BANDWIDTH (MHz)
L	1.215 - 1.3	85
S	3.1 - 3.3	200
C	5.25 - 5.35	100
X	9.5 - 9.8	300
J	13.4 - 14.0	600
Ka	35.5 - 35.6	100

Table 2b Bandwidth Allocations

Platform

Structure	150
Thermal Control	20
Unified Prop Sys	50
AOCS (using CMGs)	60
TT&C	15
Solar Array ($1m^2$)	15
$NiCd_2$ Batteries (500Whr)	40
Bus Electronics	10
Sub-total	360 kg

Downlink

Transmitter	15
Antenna	5
Sub-total	20 kg

SAR Sensor

Active Antenna ($6m^2$)	200
Central Electronics	75
Sub-total	275 kg

Overall Satellite Mass

Platform	360
Downlink	20
SAR Sensor	275
TOTAL	655 kg

Table 3 TACSAT SAR Dry Mass Budget

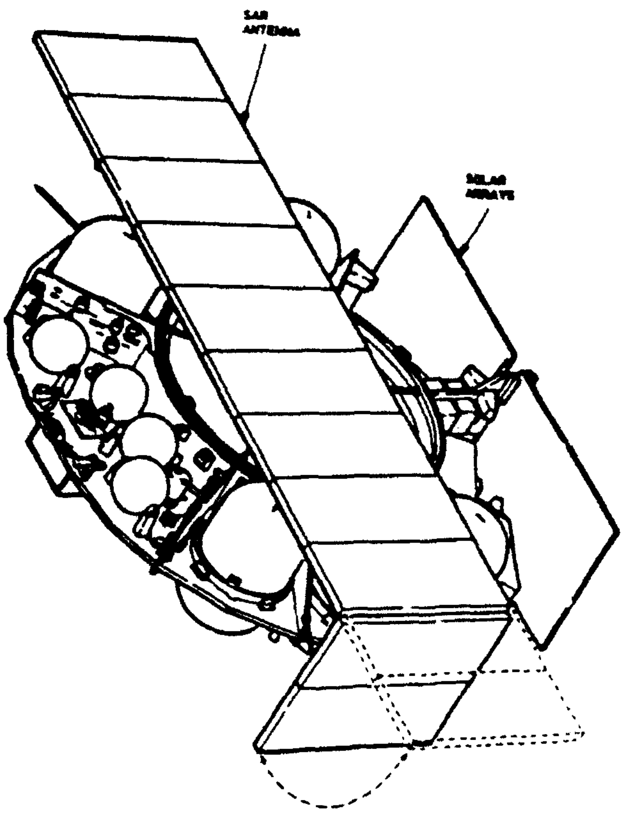

Figure 1 **TACSAT SAR Configuration**

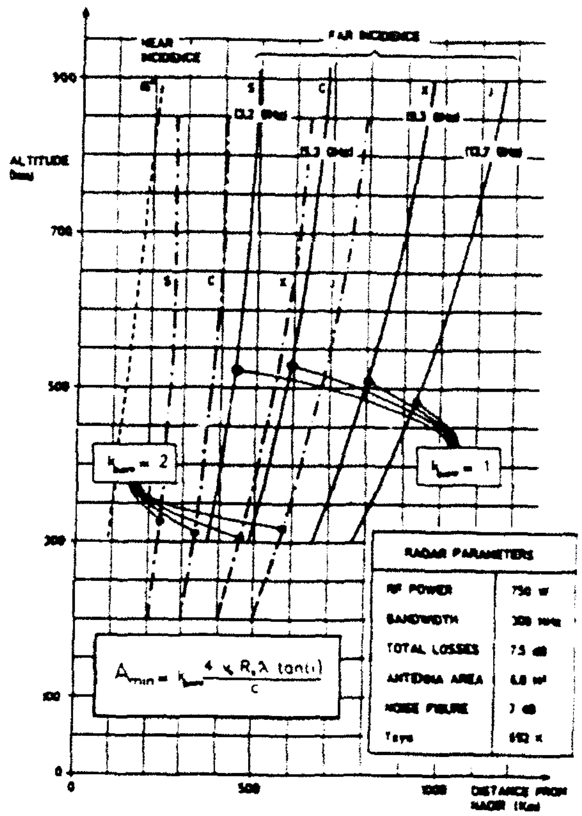

Figure 3 **Orbit Altitude versus Access Coverage Width**

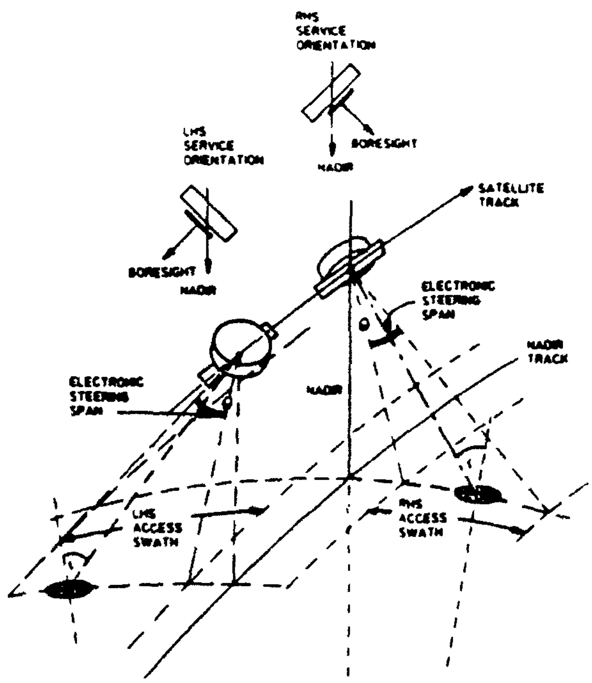

Figure 2 **Operating Geometry**

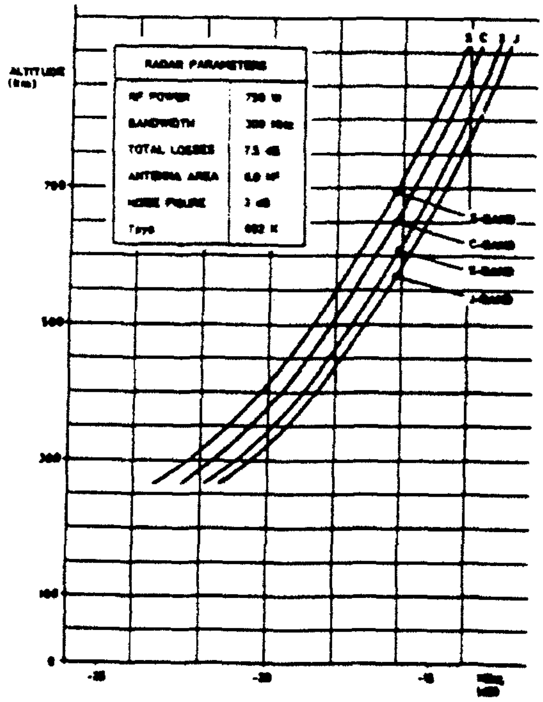

Figure 4 **Orbit Altitude versus Worst Case Sensitivity**

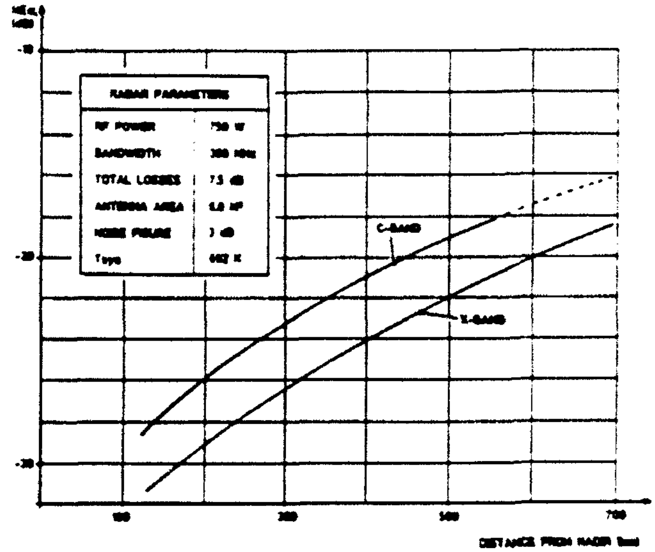

Figure 5 **Sensitivity versus Distance from Nadir**

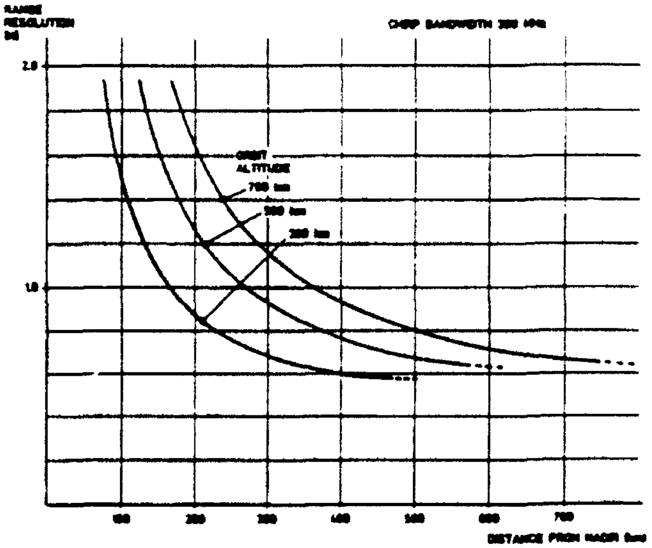

Figure 6 **Slant Range Resolution versus Distance from Nadir**

LOW COST TACSAT FOR NATO SURVEILLANCE

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SUMMARY

Cost and performance technology breakthroughs in recent years have made possible the near term development and deployment of a low cost TACSAT surveillance system. The potential for both the spacecraft and sensor to be launched on lower cost launch systems is the fundamental basis for reduced system cost. Additionally, the spacecraft and sensor must be amenable to low cost mass production. For all weather, day/night surveillance, an active synthetic aperture radar (SAR) is the ideal instrument, and the ability to mass manufacture thousands of very low cost transmit/receive (T/R) modules has been developed at Hughes. Since the size of a SAR increases as the orbital altitude is increased, a final element of the economical basis for a low cost TACSAT is the definition of low earth orbit (LEO) space constellations with high mission utility. Tradeoffs are also provided to compare sparse or zero based peacetime constellations with surge launch capabilities to small peacetime early warning space constellations that can also instantly provide regional combat support on demand.

1. INTRODUCTION

With the decrease in the long range ICBM strategic threat and the increasing likelihood that theater or regional engagements are the more probable form of conflict, local and battle area surveillance ascends to paramount importance.

Such missions as regional treaty monitoring and verification, area observation to detect military buildup and movement, assessment of force strength, characterization of deployment elements, and battle damage assessment must be addressed. A major added operational requirement to accomplish these missions is that they be performed under all weather conditions and during the day and night. Numerous trades have shown that such observations are typically beyond the reach of organic assets, including both ground and airborne systems, and therefore would best be satisfied by spaceborne sensor platforms.

2. ENABLING TECHNOLOGY

The confluence of two significant events—namely the evolution (and recognition) of the regional missions described above, and the maturation of the essential enabling technology—has made possible the near term satisfaction of these mission shortfalls. As is suggested by the scope of this symposium, the solution lies in the development and deployment of LEO spacecraft and sensors that can be produced and launched at low cost. Analysis has shown that the major way to economize on space systems is to develop ultra-lightweight spacecraft and sensors to facilitate their launch on inexpensive launch systems. Many low cost booster systems are already available to place lightweight spacecraft into LEO: Pegasus, Taurus, Titan II, Delta II, Atlas, Delta, Ariane, etc. Thus, whenever a space system requires multiple launches, such as to replace relatively short lived spacecraft/sensors by the insertion of new technology, or to establish a small constellation of peacetime satellites that can accomplish both early warning and battle support missions, the greatest single life cycle cost saver is reducing individual launch costs.

2.1 LAUNCH COSTS AND ORBIT ALTITUDE

As stated, the major factor in achieving low cost tactical surveillance is in the recurring cost of the launch system. However, the cost is directly dependent on the payload weight and the characteristics of the orbits into which the payloads must be placed. For radar, for example, as the sensor is placed into higher and higher orbits, the size of the radar (often referred to as the power aperture product) increases the same as the range to the fourth power. Thus, at the higher orbital altitudes, an active (or even a passive) sensor will be much greater in size and weight. Since spacecraft payload/launch costs vary directly to size and weight, and it is therefore much more economical to launch lighter payloads into LEO, the remote sensing functions tend to be most cost-effective at LEOs. However, a countervailing issue to be dealt with is the number of LEO spacecraft/sensors required to provide the necessary viewing time or coverage of the region or theater in question. It is a truism that as the orbit altitude of satellites is reduced, the number of spacecraft required to maintain the same ground viewing coverage increases. Tradeoffs that can compromise the conflict between orbit altitude and spacecraft constellation size are presented later.

2.2 SENSOR TECHNOLOGY AND COSTS

For compatibility with the smaller, more economical launch system architecture, the surveillance sensor must be ultra-lightweight. But another important enabling technology for the sensor is that it be amenable to low cost mass producibility. Specifically, the productization design must lend itself to automated, non-person-intensive production and test. A description of the way this has been accomplished at Hughes for spaceborne radars is presented later.

3. LAUNCH-ON-DEMAND VERSUS PEACETIME CONSTELLATION

Two ways of carrying out regional or theater surveillance are obvious. The first would rely on nonspace resources to ascertain that elevated threat levels are imminent or have occurred. The second would rely on the peacetime global reach of space to monitor and provide the earliest possible warning that a threatening situation is developing. Also, a hybrid approach that has no or a single surveillance spacecraft on orbit and that provides for surge launches of additional sensors to support the execution of theater battle is feasible.

The above dichotomy has very important implications for the design and operational characteristics of the spacecraft and surveillance instrument. Hughes has performed analysis of a launch-on-demand space system that would be developed with the following operational scenario in mind.

3.1 LAUNCH-ON-DEMAND OPERATIONAL SCENARIO

To satisfy a growing need for dedicated and available space support for tactical military commanders, the future space systems might represent a significant departure from the way current space systems are designed, built, and deployed. The advanced spaceborne radar would be designed for quantity production, long term storage, and easy checkout, such as the state-of-the-art missile systems built for years at Hughes. A surprisingly close match between the content of certain missiles developed by Hughes and spacecraft was discovered. Specifically, the percentages of the total units in terms of structure, payload, and propulsion were remarkably similar. Many of these missiles were developed to be used in much the same manner, including long term storage in the field, rapid go/no-go checkout, and launch. As shown in Figure 1, a rather small radar satellite would be compatible with stowage in, for example, a Pegasus fairing (a) and would be designed for low cost, quantity production (b). The radar satellite would be capable of long term field storage (c) so that a ready-to-launch inventory would always be available (d). Built-in self-test capabilities would permit rapid checkout of the radar spacecraft by the use of simple support equipment in the factory and the field. Launch would be from a Pegasus (e) or other suitable launch system (f).

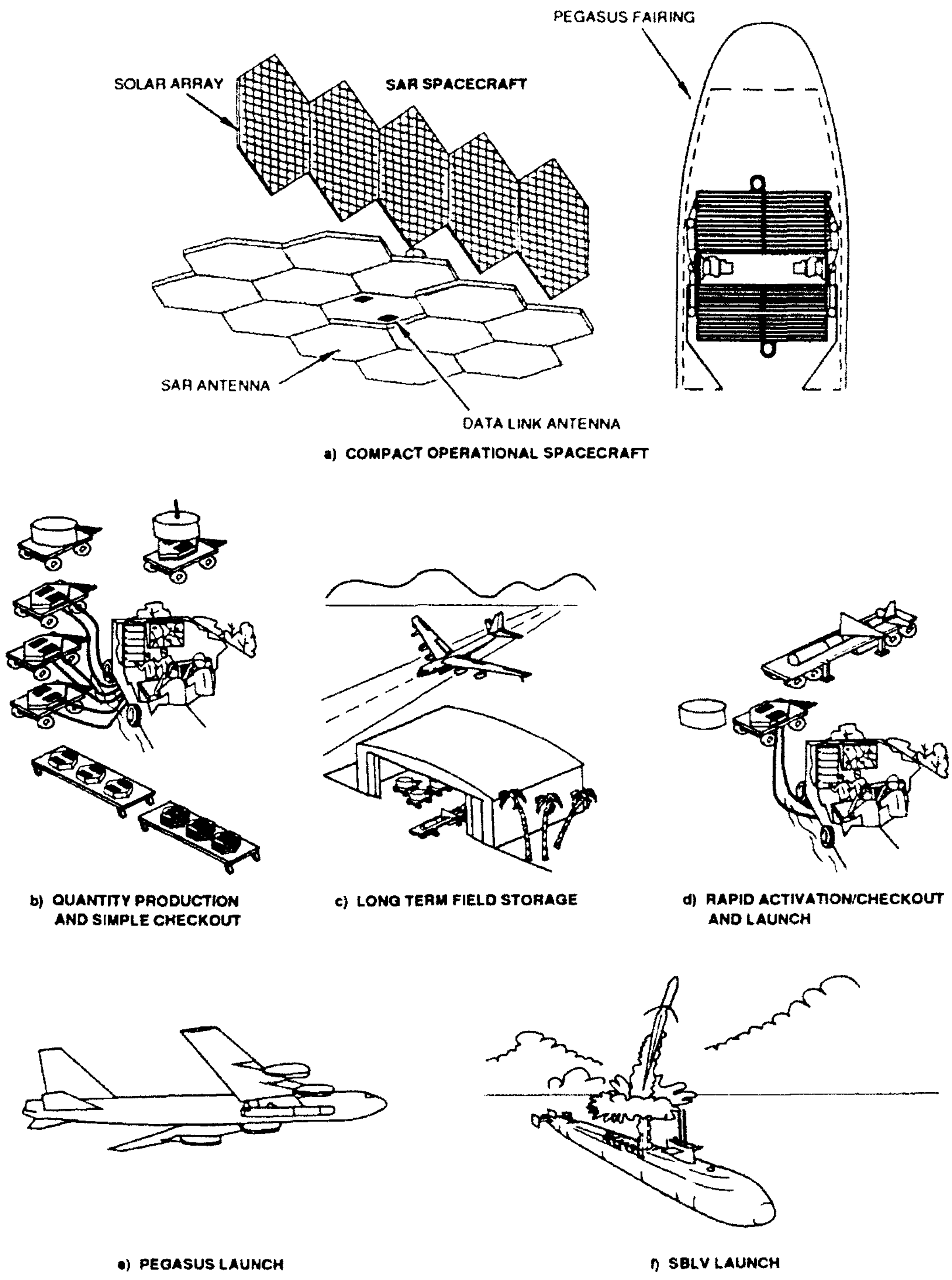


FIGURE 1. SAR SYSTEM CONCEPT

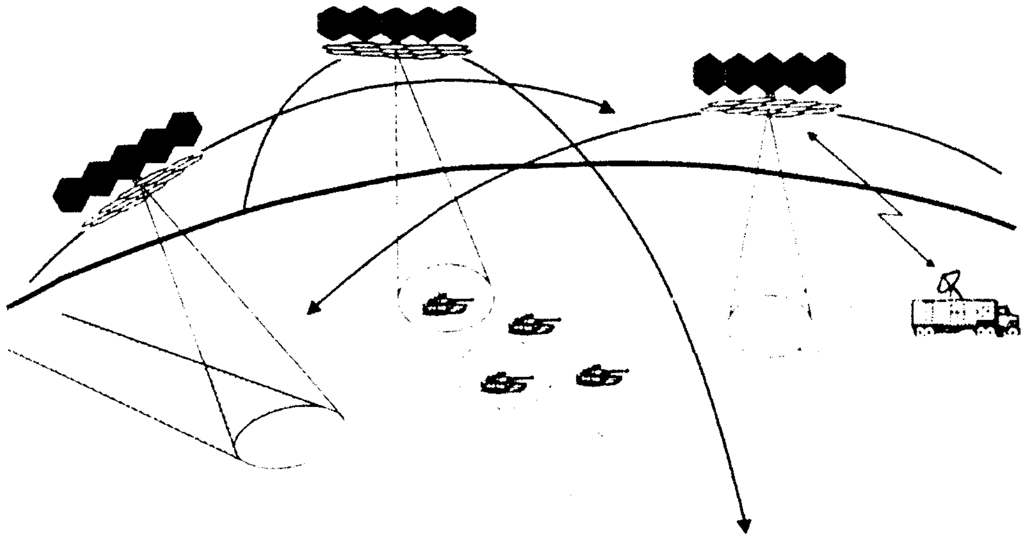


FIGURE 2. REPRESENTATIVE OPERATIONAL SAR SYSTEM

3.2 PEACETIME SPACE CONSTELLATION SCENARIO

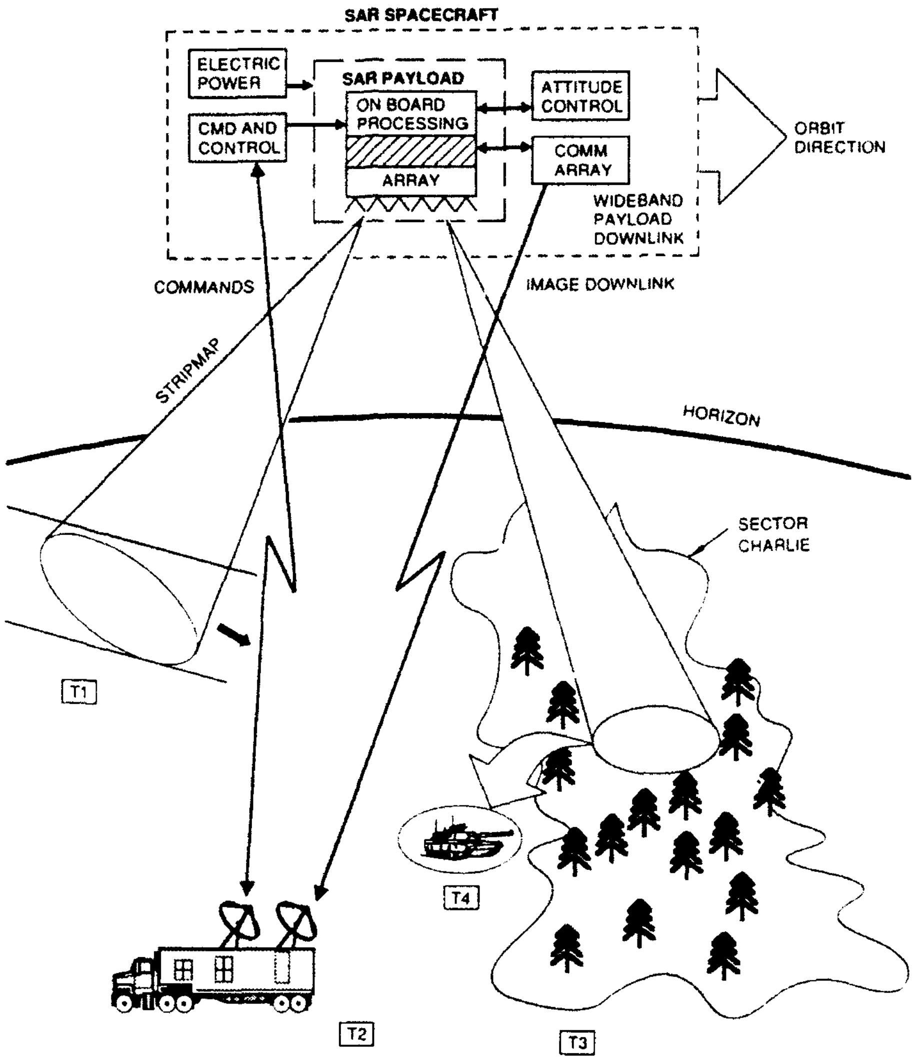
Contrary to the launch on-demand scenario, the peacetime spaceborne radar scenario makes use of a small LEO radar constellation that provides broad area theater search, as well as high resolution imaging upon the detection or cueing of a target's presence. This scenario does not preclude the capability to execute a surge launch to supplement a peacetime constellation or to reconstitute the system as satellites are destroyed during major conflicts. The control of this situation would reside in the theater commander who also has the responsibility and capability for real time tasking of the spaceborne radar and spacecraft. In Figure 2, three radars operating in the SAR mode are depicted, one providing strip map imaging in a wide area target detection mode and the other two providing the higher resolution spotlighting to assist in characterizing enemy targets that have been detected. Mode change instructions to look in different areas, for example, are commanded in real time by the field commander's mobile control station, which can also accept onboard processed or partially processed imaging data from the orbiting radars. Depending on the time frame of the first launch, there is an important tradeoff between accomplishing all imaging processing onboard the spacecraft, including comparison of input data to stored images of potential targets, versus the downlinking of rather wideband imaging data for near real time processing in the commander's ground control console. The latter is most likely to be preferred, until completely reliable autoprocessing of imaging data has been successfully validated. The implementation of an efficient wideband downlink is very straightforward and can even use the onboard radar array to provide the communications downlink aperture so that another transmitter/antenna would not be necessary.

The provision of a commander's command control console and work station to task and control the spacecraft and sensor, process collected data, and provide data fusion as required is an essential part of the total operational system, along with the wideband real time downlink into these ground stations.

A typical operational timeline has been developed to demonstrate how the system might operate. As shown in Figure 3, as the spacecraft rises above the battle area at time T1, it begins its preprogrammed broad area strip map search for enemy targets. At time T2, the mobile ground station with a small antenna transmits a command for the SAR to "spotlight" a portion of Sector Charlie where unrecognized enemy assets may be hidden in a wooded area. At time T3, the SAR spotlights Sector Charlie at high resolution, using a scanning beam, and accomplishes repeat looks as necessary to increase resolution. At time T4, a SAR processed or partially processed image is downlinked to the mobile ground station, which reveals a cluster of enemy tanks.

4. COST AND PERFORMANCE TECHNOLOGY BREAKTHROUGH

For several decades Hughes has developed space based radars that have performed with distinction. One of the earliest was Pioneer Venus in 1978, which was the first spacecraft to map Venus. This thorough mapping was exceptionally revealing about the planet, despite the fact that the SAR resolution was approximately 100 km. Subsequently, Hughes developed the Magellan radar, which mapped Venus at a considerably higher resolution — 100 meters. All U.S. space shuttles have flown a Hughes radar and communications system, which has been used successfully and repeatedly for space docking and spacecraft retrieval maneuvers. A look at the Magellan spacecraft (Figure 4) shows that the SAR functioned with a conventional parabolic antenna and amplifier tube and that its performance was limited to the kinds of power/amplifier products available at the time it was built. Figure 5 shows an architecture and performance attribute comparison between the Magellan architecture and the Hughes-built radars of today. Note that the reflector system uses the conventional, gain limited tube amplifier and comprises a single black box transmitter and receiver. A major shortcoming of this architecture is the poor beam agility, which can be accomplished only by mechanically steering the antenna for any beam steering or scanning mode of operation. The phased array architecture on the bottom of the figure shows the ability to construct and deploy very large arrays with thousands of low power T/R channels that can provide essentially limitless performance. When this architecture is used in a space communications mode, for example, it means that very small, low power ground terminals can be deployed to operate with this spaceborne aperture because of its potentially large directivity and EIRP. Perhaps the single most important characteristic, however, is that the phased array provides for instantaneous beam



LEGEND

T1 SAR SPACECRAFT COLLECTING BATTLE AREA STRIPMAP IMAGES
T2 MOBILE TACTICAL CONTROL COMMANDS SAR TO SPOTLIGHT SECTOR CHARLIE
T3 SAR SPACECRAFT SPOTLIGHTS SECTOR CHARLIE
T4 SAR SPACECRAFT DOWNLINKS SPOT IMAGE TO MOBILE TACTICAL IMAGE

FIGURE 3. TIMELINE FOR 7 MINUTE SAR FLYOVER

scanning in all directions. Typical applications have provided for off-boresight scanning in all directions of greater than 60°. A principal physical attribute that is essential to the NATO architecture of the future is that these large phased arrays have a very low profile and extremely lightweight densities—approximately 7 kg/m^2 or less.

Spaceborne phased array radars for military applications have not existed for one reason: cost. The major cost item has been the T/R module, since a given radar application may require thousands or even tens of thousands of these devices. Hughes has made enormous investments in T/R modules that were primarily directed at airborne applications, since Hughes has been a major airborne military radar supplier for many years. These T/R modules incorporate monolithic microwave integrated circuits (MMICs) that are mass produced, weigh approximately 2 grams (depending on the frequency band), and are typically packaged in a 1 x 0.3 x 0.11 inch unit. Low temperature cofired ceramics (LTCCs) have been used for several applications. LTCC is a technique in which active and passive components are deposited within ribbons of ceramic, and the entire package is heated at approximately 600°C lower than the temperature ordinarily used on ceramics, so that the total package is formed at once without damage to the functional devices. When this process was begun years ago, a single channel T/R typically cost several thousand dollars; today these devices are available at less than $500 each. In addition to this, Hughes, under a U.S. Government Manufacturing Technology (MANTECH) program, constructed an automated (robotic) facility that is dedicated to the mass, low cost production of T/R modules for radar applications. This facility, which was newly constructed and dedicated early in 1991, is being further automated each year, and the cost of T/R modules by 1994 will be between $200 to $400 each. This represents an order of magnitude reduction in cost from a few years ago. Refer to Figure 6.

5. REPRESENTATIVE ORBIT CONSTELLATIONS

A wide variety of low cost, high performance space constellations can be devised. One small constellation would be

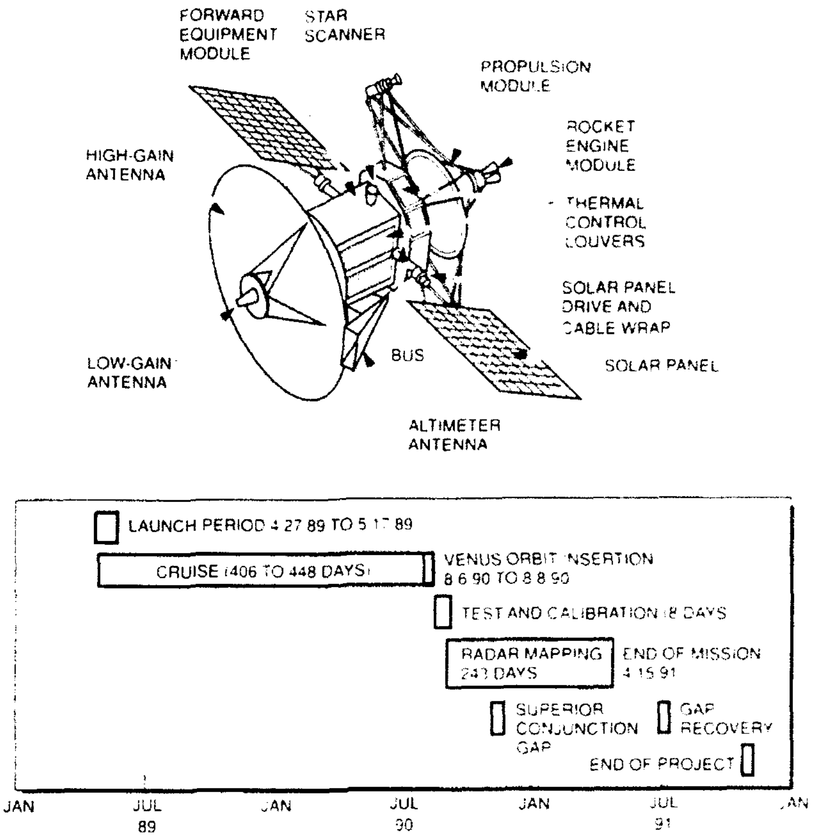


FIGURE 4. MAGELLAN SPACECRAFT

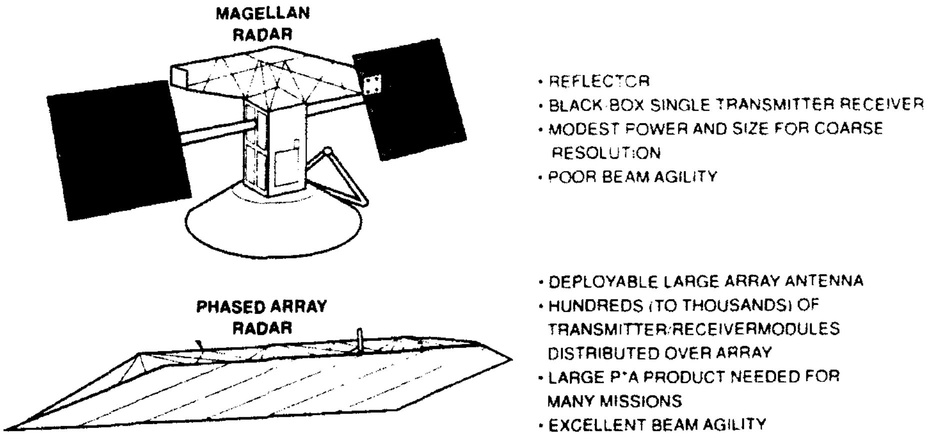


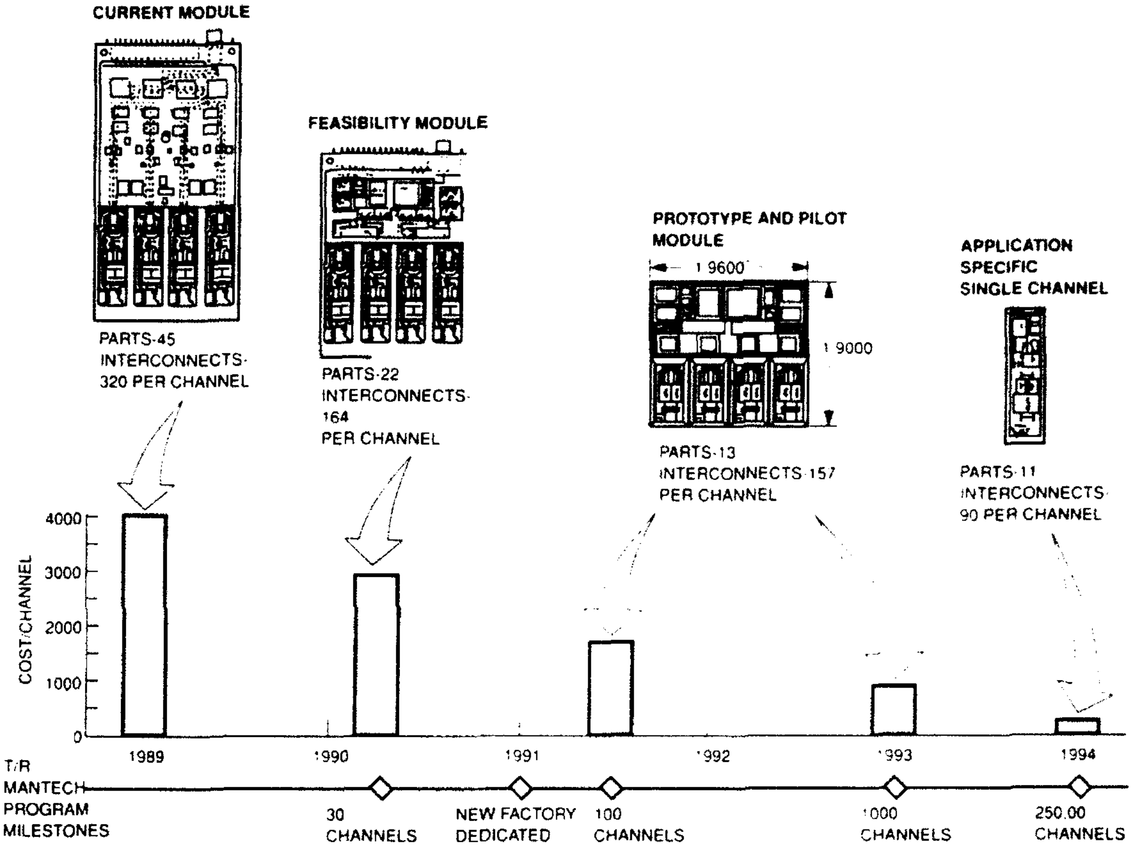
FIGURE 5. RADAR ARCHITECTURE COMPARISON

placed at an orbital altitude of 600 n.mi. (see Figure 7). If truly global surveillance coverage is the goal, including excellent coverage over the poles (which is not the most likely scenario for the majority of the NATO theater surveillance missions), a system of four satellites comprising two planes with two satellites each, all with a 90° inclination, would be feasible. The satellites would be separated by 180° within each plane. The global coverage would be very good in that the worst case revisit gap in the entire world (meaning the time during which no sensor can "view" that particular area on the earth) would be 1.54 hours (and this would be a rare occurrence). However, the average revisit gap for every place on the planet would be 0.62 hour (approximately 37 minutes).

Another four-satellite constellation would be placed at half the altitude of the one just described (see Figure 8). In this case, the altitude for all spacecraft would be 300 n.mi., and the inclinations for the spacecraft would be 0°, 90°, 57°, and 57°, respectively. The orbit period for each would be 96 minutes, and the average revisit gap globally would range from 2 to 4 hours. Figure 8 shows the ground traces over three revolutions (4.8 hours) for these four satellites.

6. REPRESENTATIVE LOW COST SPACE SYSTEM DESIGN

Hughes has developed a low cost spacecraft and SAR payload that represents one form of low orbit, long life approach to the



NOTE: MANTECH MODULE EVOLUTION PROVIDES COST AND PRODUCIBILITY BENEFITS

FIGURE 6. MANTECH PROGRAM

problem. The goal was to design a lightweight, low cost, manufacturable spacecraft that could support a fairly substantial (but ultra-lightweight) SAR and be compatible with the Delta II launch system. The design is for an afternoon sun synchronous orbit with an altitude of approximately 370 n.mi. The primary solar power system is approximately 5 kW, and the SAR array is approximately 12 × 4 meters, for a total area of 48 m^2. In this particular design, it is desirable to include a large (6 foot diameter), dual gimbal pointing parabolic antenna to relay wideband SAR data to a geosynchronous communications relay satellite. Furthermore, an X-band downlink is provided to enable real time SAR data to be transmitted to operational users directly below the spacecraft/SAR at any time. The entire spacecraft, power system, radar, propulsion system, and all onboard electronics are designed for an orbital lifetime of 8 years.

The very lightweight spacecraft for this application is a simple triangular truss arrangement, whereby the SAR array is affixed to one of three sides of the bus, and each of the solar wings is attached to one of the two remaining sides, as is shown in Figure 9. Note that in this figure, the fairing above the Delta II second stage has a 110 inch envelope for the payload, and the large reflector antenna is also accommodated within the fairing.

Figure 10 depicts this spacecraft/SAR payload in the space-deployed configuration. The SAR antenna is canted approximately 30° off the horizontal, to permit side-looking

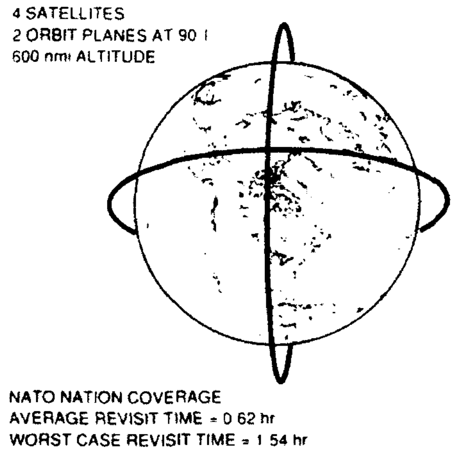


FIGURE 7. HIGH ORBITAL ALTITUDE CONSTELLATION

only. In this particular design, the requirements are for ±30° scan capability in the elevation axis and 20° scan capability in the azimuth axis. Similar designs have called for the array to be deployed normal to the earth's radius vector to more readily scan to the right and left as the spacecraft progresses along the orbital path. In the latter case, the designs have typically called for ±60° of scan in both axes.

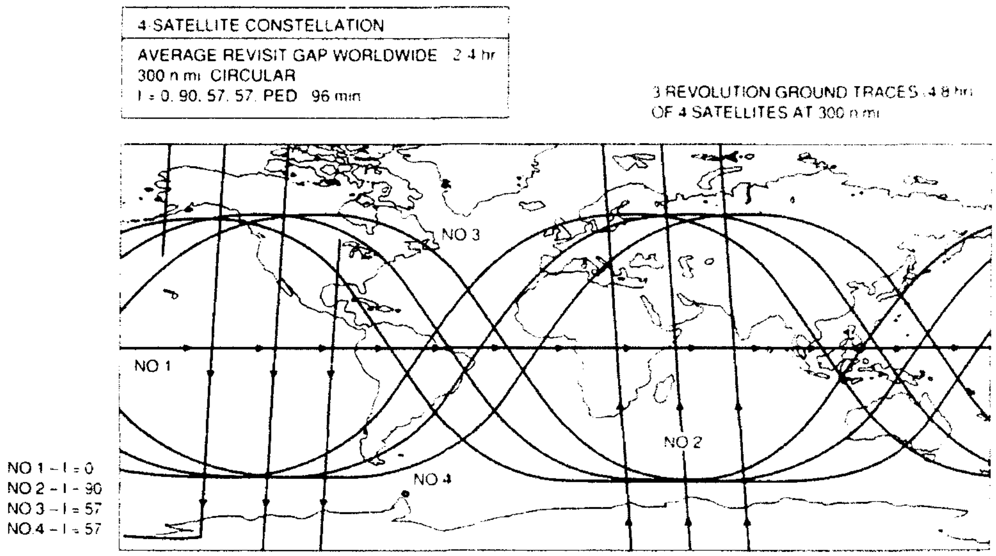


FIGURE 8. SATELLITE GROUND TRACES

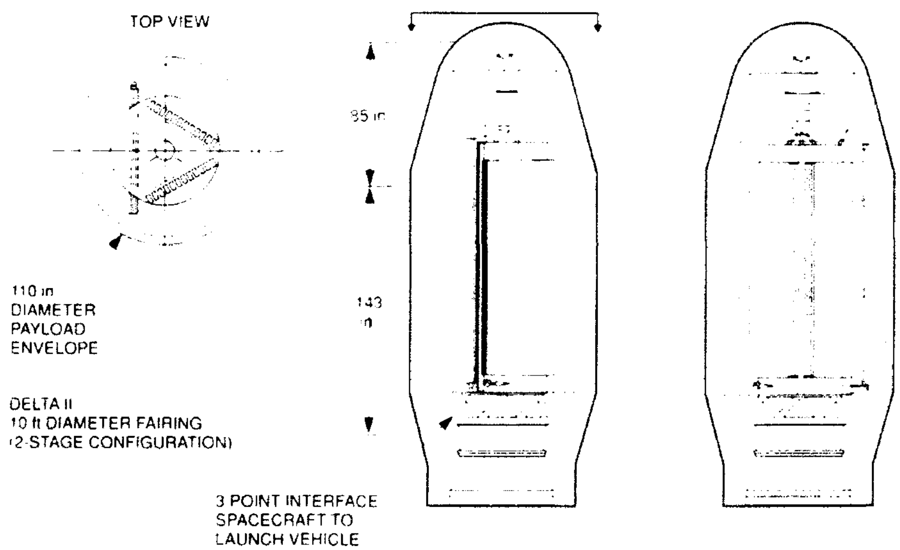


FIGURE 9. SPACECRAFT LAUNCH CONFIGURATION

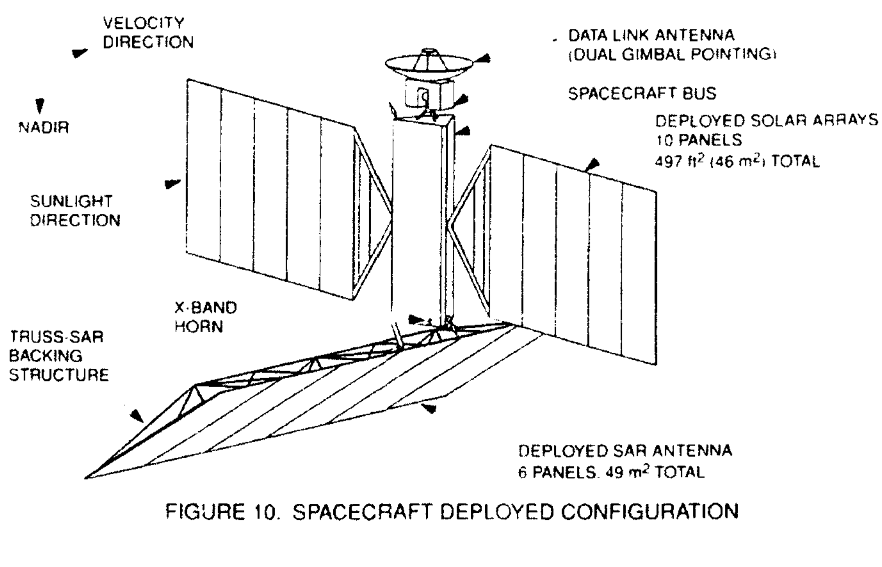


FIGURE 10. SPACECRAFT DEPLOYED CONFIGURATION

Discussion

Question: May you give (rough order of magnitude) how many PEGASUS launched small satellites you plan to build to have an economically efficient approach in terms of manufacturing facilities and effective service?

Reply: That one concept was based on a Customer long-term need for hundreds of spacecraft at very low cost to be deployed in the field for long periods. The PEGASUS solution was a design constraint, but is high in $s /lb to orbit. Multiple spacecraft launchers for peacetime constellations and short (approximately 1 year) orbit life may make more sense. However, the very large, high performance radar (on Delta II) is close in cost to the TACSAT goals discussed by Charlie Heimach, et al.

Cost Effective TacSats for Military Support

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Abstract

This paper discusses the key space system features required to provide effective support to military field units and discusses Lockheed's concept to fill critical niches in military surveillance and remote sensing.

The military is aware of benefits available from more effective use of space. To provide those benefits, space systems must be dependable, easy to use, flexible, responsive and affordable. This paper describes how a programmable, multi-spectral E-O sensor combined with a new small satellite bus, and supporting command and control and data processing tools can provide environmental and surveillance support to users in peacetime, crisis and combat. Also discussed are how such a system meets essential military utility criteria and an approach for rapid TacSat system development, evaluation and deployment.

Introduction

In an era when businesses are learning the importance of total customer satisfaction, developers of new space systems have opportunities to achieve similar goals. Just as products emphasizing user convenience, hassle-free service and high quality at an affordable price are essential in today's economy, space system developers must use emerging technologies to meet similar needs for civil and military systems. In the commercial arena, a system designed for total customer satisfaction, Motorola's IRIDIUM$^{TM/SM}$ global personal communications system is in development. The same philosophies can be reflected in a tactical surveillance system and the projected commercial infrastructure can aid the economic feasibility of such a system.

The IRIDIUM system provides global personal communications using a constellation of small LEO satellites, supporting ground facilities and a user equipment family including a small, hand-held telephone and a variety of data terminals. The system thrust is to provide cellular telephone-like convenience and quality with 24 hour dependability anywhere on the face of the Earth. The IRIDIUM system will provide high quality telephone connections anywhere with the same effort normally required to call between offices in a large city. In short, the IRIDIUM system applies state-of-the-art technologies to make things simple and convenient for customers. With modern cryptographic devices and GPS receivers, an IRIDIUM subscriber unit could be ideal for several voice and precision location functions in peacetime and limited warfare situations. Another military benefit from the IRIDIUM system may be the user satisfaction model it provides. A like philosophy, applied to the development of military surveillance systems, could produce huge benefits.

Any discussion of a new military capability must address new world realities. The welcome reduction in superpower tensions has come with unpleasant side-effects including greater political instability, local "hot spots," and the danger of a local conflict spiraling out of control. The political reality of reduced western military spending and constrained forward basing accompanies these threats. The factors combine to demand greater flexibility and responsiveness of the remaining forces. The Persian Gulf showed that timely warning and intelligence information in the hands of commanders can produce force multipliers that allow the prosecution of conflict with low casualty levels.

One consequence of this environment is an emerging need for user controlled, cost effective, tactical space systems to provide timely information to commanders. The systems could give indications and warnings of force build-ups, allow treaty compliance monitoring, and support national and allied forces in peacetime, crisis and combat. They could operate in modes to satisfy the different needs of several users.

The needed capability can be effectively provided by a small LEO satellite with a programmable, multi-spectral sensor combined into a "TacSat" space system shown in Figure 1. Key TacSat system features would include a distributed architecture with direct user tasking and data readout. Technologies central to this TacSat system concept are multi-band, CCD focal-plane arrays, satellite on-board computing and advanced work station ground data processing.

Mission Requirements & Trades

User Satisfaction

At the most basic level, the fundamental requirement of any system is that it satisfy user needs. In a military surveillance system, that translates into a need to provide timely information on force disposition, local environmental, infrastructure and other conditions to commanders. It is technically feasible to build a space

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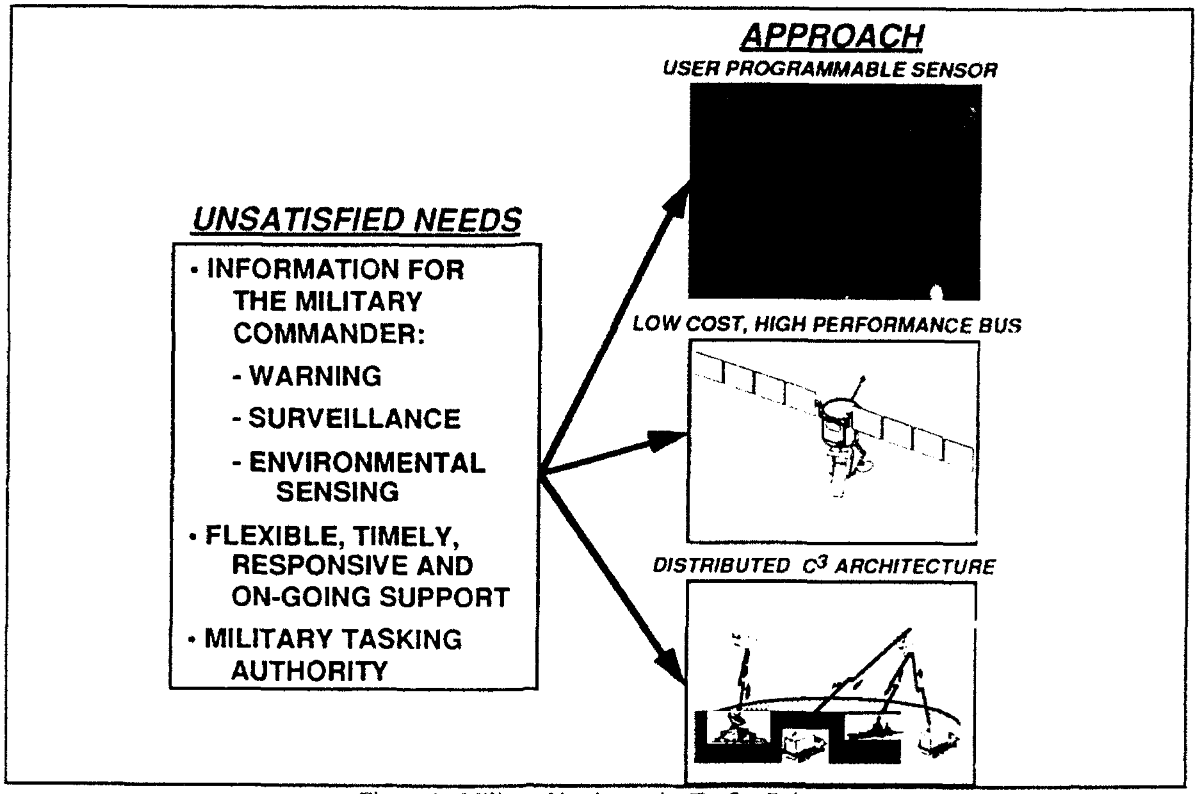


Figure 1. Military Needs are the TacSat Driver

system with the spatial and spectral resolution to do the job. It is harder to retain adequate resolution while satisfying frequent coverage, large area surveillance, timely access to information products, ease of use, and dependability needs, all with low system costs. Many think that the simultaneous satisfaction of these needs is not practical. It is practical, however, to build a system that meets these varied user needs at selected times. Using low cost systems and technologies, it is possible to achieve high user satisfaction with a robust and responsive system architecture. De-centralization does not result in higher costs than current capabilities. This architecture, shown in Figure 2, manages resources to fit a situation and is the thrust of our TacSat concept.

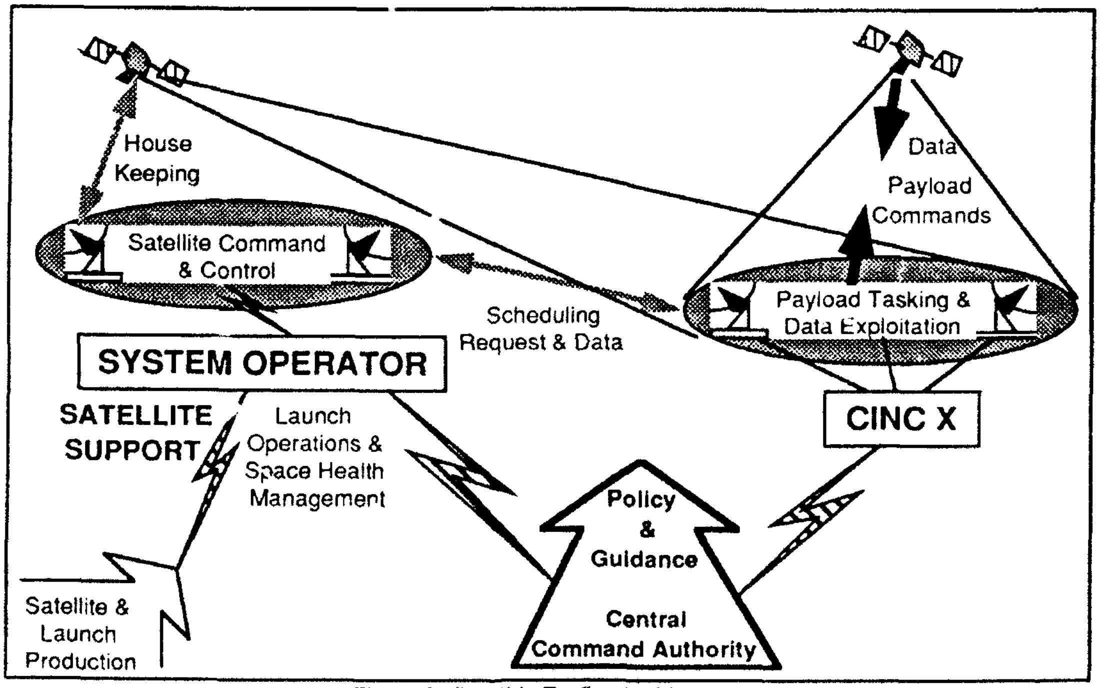


Figure 2. Possible TacSat Architecture

Operational Characteristics	Multi Mode Sensor	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
Imaging Capability	X							
Day/Night Utility	X	X						
Non-Visual Capability	X	X						
Flexible Operation	X	X						
Large Access Area			X	X				
Timely Access to Info			X		X	X	X	
Ease of Use		X		X		X		X
Dependability				X	X	X	X	

Figure 3. User Driven Design Feature Matrix

Figure 3 shows the key user needs satisfied by an electro-optical TacSat system, and how those needs map into desired design features. The first system design step is the definition of key mission driven sensor characteristics. These characteristics, combined with orbit selection, define the system's "capability to survey and collect".

System Performance

TacSat system performance is fundamentally defined by the combination of sensor characteristics and mission orbit. As shown in Figure 4, the process of selecting these parameters starts with the definition of mission requirements; i.e., what specific roles will the system be designed to satisfy and what performance levels will be required for these roles. For example, one role might be daylight surveillance of a battle area to detect armor, another might be surveillance of a port area to detect shipping activity. Each of these missions requires some specific quantifiable threshold performance and a range of characteristic goals that affect tasking. The key to sensor and orbit design is balancing as many tasking thresholds as practical while maintaining performance against the variable goals, all at an affordable cost.

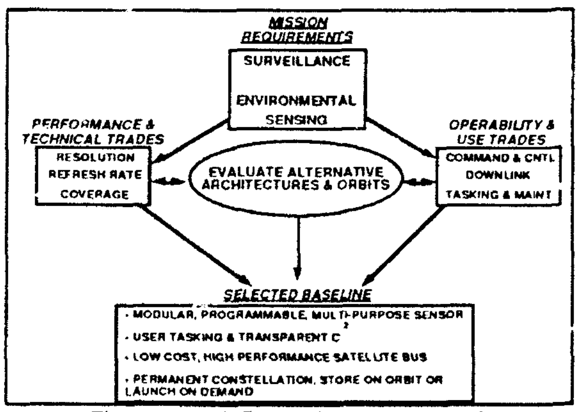


Figure 4. E-O Sensor System Approach

The multiple areas of performance required for TacSat missions are achievable in several ways. The conventional approach is to use two main sensor types, one emphasizing wide field-of-view, the other high resolution. **The key to our concept is the capability to satisfy the key TacSat missions with a single sensor.**

This is possible through the use of two different focal planes and the flexibility to operate in either "staring" or "scanning" modes. The "staring" mode captures high resolution data of a relatively small geographic area. In this mode, a sensor continuously tracks, or "stares" at a particular target, and the image size is limited to the sensor's field of view (FOV).

The sensor can scan in either the "push-broom" or "whisk-broom" modes. When scanning in the "push-broom" mode, a sensor images a strip determined by its FOV. A wider area, with reduced dim target performance, can be imaged using the "whisk-broom" scan mode. This dual-mode capability, discussed later, provides a surveillance option "menu" to the user ranging from small area, higher resolution to large area, medium resolution. In addition, the pointing capability required for scanning and staring also allows target area selection within a field of regard (FOR) of about ±45° to each side of nadir along the flight path.

In addition to user-programmable pointing, the sensor should feature multi-spectral capability. The scan modes might be supported by a series of multi-element linear focal-plane arrays, each covered by narrow band optical filters. The high resolution staring mode could use a square panchromatic array operating in the visible band. This combination makes it possible to support the many spectral bands and differing resolutions needed to perform TacSat missions.

Coverage

TacSat system flexibility and timeliness are, to a large extent, orbit dependent. Of course, orbit selection often involves trades against resolution. Orbit selection requires consideration of options for altitude, inclination and, in some cases, eccentricity. Figure 5 shows some of the factors effecting altitude selection.

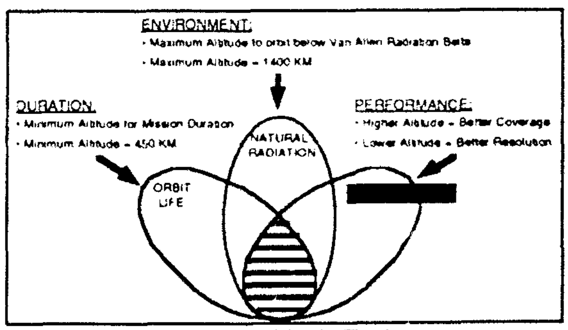


Figure 5. Altitude Trades

Environmental factors bound the range of available options. Below about 450 KM, atmospheric drag starts to become a factor by increasing propellant for orbit maintenance and beginning to produce attitude control design issues. At the other extreme, altitudes above about 1400 KM result in increased requirements for component hardening against the radiation environment. (Orbits above the radiation belts would not need as much hardening, but would require sensors of a different class than those discussed here.)

Specific altitude selection between 450 and 1400 KM involves minimizing the total system cost for a desired resolution level. As altitude increases, sensor focal length must also increase to maintain constant resolution on the ground. Increasing the focal length usually increases both the sensor and satellite weight. This combination of increased weight and altitude increases satellite cost and, generally, launch costs. However, the higher altitudes have several significant advantages. First, even at constant resolution levels, it is possible to gain a wider access area on a single pass by taking advantage of greater off-nadir performance. Second, if reduced resolution is allowed, higher altitudes allow larger imaging swaths on a single pass even with a constant down-link data rate. Therefore, a sensible design strategy is to optimize for the highest practical orbit (below 1400 KM) available with the selected launch vehicle. Sizing the sensor to provide the required resolution would then give the best balance between resolution, single satellite coverage and cost. This paper assumes that booster constraints would result in a system at the low end of the range (450 KM), but no final decision is implied.

Other important parameters are whole earth versus mid-latitude coverage, revisit times and the number of satellites required to achieve them. Orbit inclination is selected based on the desired range of target area latitudes and the practical constraints of available launch sites, though some mobile launch systems could eliminate those constraints given sufficient payload capability. The trade is usually between a polar orbit and an inclined orbit, either circular or elliptical. The polar orbits have the advantage of covering the entire earth with potentially repeating ground tracks. Their disadvantage is that the satellite spends a significant portion of its orbit over the polar regions. Inclined orbits can cover the temperate zones or higher latitudes as required. However, it is difficult if not impossible to design an inclined orbit where a target is available at the same time each day with only a few satellites.

Per satellite coverage is a key measure of cost effectiveness and is, in turn, driven heavily by launcher options in two ways. First, use of maximum launcher performance increases orbital altitude and coverage. Second, launcher flexibility allows varying, situation specific orbits when appropriate. Another approach is a general purpose sun-synchronous, circular, near polar orbit that provides a global, long term capability.

System Useability and Operations

An effective command and control architecture is the key to TacSat system useability, as shown in Figure 2. The primary architectural strategy divides control into two functions, mission support and satellite support. **The clear separation of these two functions is critical to user satisfaction since it allows combat commands to obtain satellite surveillance benefits without the burden of satellite management.** This support function, completely transparent to the field commands, is accomplished by rear echelon activities with the facilities and expertise to manage and control a single TacSat or a multiple-satellite constellation.

The operational principle is based on the idea of field commanders using a predetermined schedule of resource availability to initiate targeting. They would know in advance of satellite orbital passes over their areas of interest. The rear echelon activity would operate the spacecraft and perform housekeeping functions. Users would know that, between specified times, they could point the sensor, select spectral bands to acquire needed data and direct it to the appropriate user ground station. They would not have to control satellite attitude or provide pointing angles. Instead, they would provide target coordinates. System design features and user task planning tools would ensure that the user commands were within system capabilities, and not potentially damaging to the satellite.

The strategy allows regional commands to use a satellite as an organic asset when it is overhead, even though LEO satellites are inherently inorganic global assets. Users would control the assets based on local priorities. It would be possible for a command to make a decision on the best tasking use just prior to a pass, send commands to the TacSat, have it image the target, and return the data for processing and analysis. This high degree of user autonomy requires that the satellite and support system be designed from the start to separate housekeeping and payload operations. This feature requires a rear echelon support capability that operates the satellites and provides user support. Satellite operations include such traditional functions as: launch control, injection, check-out, navigation, orbit adjusts and maintenance. User support includes the distribution and prioritization of tasking allocations, and the resolution of operational problems.

Figure 6 shows a typical TacSat pass. With the 3° FOV multi-purpose sensor, a single pass swath width is limited by the allowable sensor obliquity. At 45°, the swath is about 1040 KM by 1930 KM at 10° elevation angle. The user access area is limited by line-of-sight from the satellite to the mobile exploitation sites that receive the data. Reaches of 780 KM are conservative and could be increased by reducing allowed elevation angles or swath lengths. The linkup is made between 3° and 5° elevation and commands and other data up-linked between 5° and 10° elevation.

TacSat operational objectives to maximize user benefits and flexibility and minimize user performed satellite operations drive the C^3 architecture. Distributed payload tasking is a key to the TacSat concept. Users require a system that simply and responsively supplies support without being hampered by insufficient satellite knowledge or the burden of additional infrastructure. Figure 7 shows the C^3 capabilities divided into categories. Payload tasking consists of simple, user commands on a pre-allocated, time share basis. Allowing users to task the sensor payload in ground coordinates, eliminates the need for user ephemerides and pointing understanding.

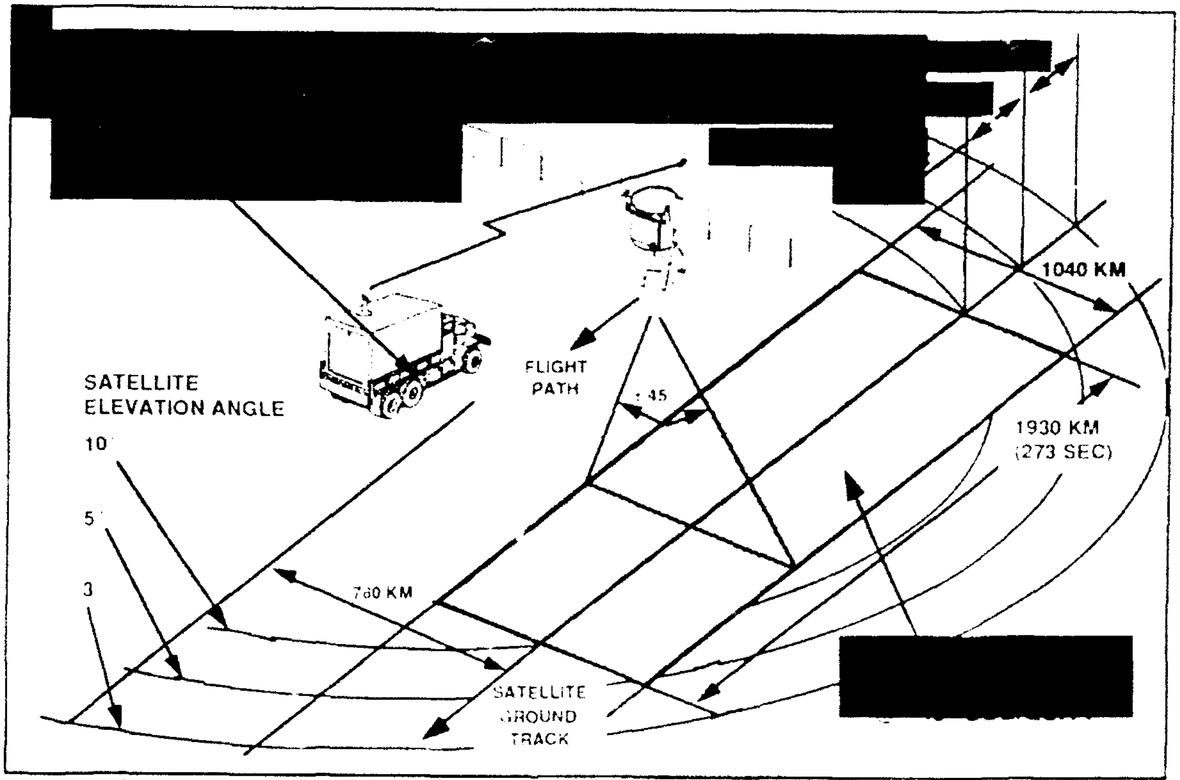


Figure 6. TacSat Operations Provide Users with a Surveillance Reach of 10[illegible] Kilometers

Function	Comments
· Satellite Control	· Centralized · Launch: Existing DOD Resources · On-orbit: Dedicated TacSat MCC
· Payload Tasking	· Decentralized · Theater Commands User
· Payload Data Distribution	· Direct To Theater Commanders Users · In Theater Distribution & Processing

Figure 7. Possible C3 Function Distribution

TACSAT Sensor

Sensor Trades and Design

Figure 8 shows a sensor system trade of sensor scan techniques. For many pointable satellite sensors, a scan mirror has been the lightweight way to cover the required field of regard (FOR). As FOR and resolution increase, this option becomes less attractive. Since the sensor operates at many wavelengths, it needs all reflective optics to maintain an accurate surface figure. A mirror large enough for the desired aperture and FOR must be stiff and thermally controlled to maintain its figure. This makes the gimbaled sensor a lighter weight design overall, though it requires a larger moving mass.

CRITERIA	SCANNING MIRROR	GIMBALED TELESCOPE
WEIGHT	LARGE EXTRA OPTICAL ELEMENT AND RIGID SUPPORT STRUCTURE REQUIRED. MIRROR NEEDS PRECISE SURFACE FOR HIGH RESOLVING POWER. ITS MASS PROVIDES STIFFNESS AND THERMAL BALANCE	HAS GREATER MOVING MASS. REQUIRES LARGER GIMBALS THAN SCAN MIRROR
SIZE	LARGER STATIC ENVELOPE TO ACCOMODATE SCAN MIRROR. DYNAMIC ENVELOPE LIMITED BUT IS AT LIMIT OF STATIC	SMALLER STATIC ENVELOPE. LARGER DYNAMIC TO ACCOMODATE SENSOR SCANNING
INERTIA AND ANGULAR MOTION	LARGE MOMENT ARM. MORE DIFFICULT TO CENTER	MUST MOVE THROUGH TWICE THE ANGLE OF A SCAN MIRROR
THERMAL	REQUIRES SUN SHIELD OR COMPLEX THERMAL CONTROL TO PREVENT SURFACE DISTORTIONS	NO ADDED EFFECT. REQUIRES NORMAL THERMAL RADIATING. SUN SHADE MAY BE REQUIRED FOR RADIATORS
SUMMARY	CUMBERSOME TECHNIQUE TO MEET COVERAGE AND RESOLUTION REQUIREMENTS	BETTER SOLUTION FOR THE VARIETY OF TACSAT MISSIONS

Figure 8. Scan Technique Trade

This is just one example of the many trades and criteria required to properly evaluate and select TacSat system design features. The sensor achieves the performance required with a single optical system serving two focal planes. It is easier to manufacture and more versatile than the alternatives, which leads to reduced costs.

Figure 9 shows a modular, multi-mode electro-optical TacSat sensor payload. The design has a wide field of view and two axis gimbals for scanning operations in both push and whisk broom modes and/or staring, as shown in Figure 10. Multi-mode operation is central to supporting different, time varying missions. With a multi-spectral focal plane, it provides IR mapping, topographic and oceanographic sensing data, and depending on objectives, could provide the bands shown in Figure 11. For higher resolution, a square, narrow FOV, visible focal plane is added.

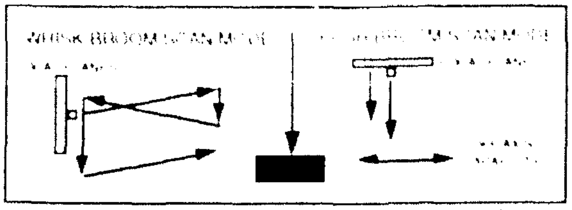
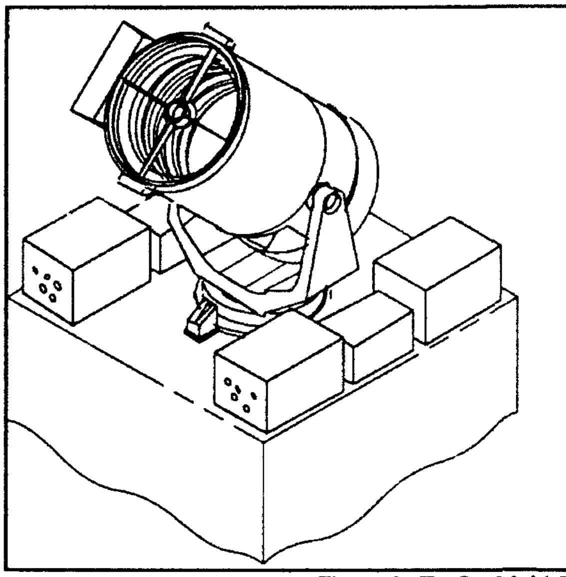


Figure 10. Alternate Sensor Acquisition Modes



• APERTURE	0.45 M
• FOCAL LENGTH	
STARING ARRAY	6.78 M
STRIP ARRAY	3.15 M
• OPTICS FIELD OF VIEW	
STARING ARRAY	.42° X .42°
STRIP ARRAY	3.00 X .1°
• POINTING CAPABILITY	± 45°
• 2-AXIS GIMBALLING WITH MOMENTUM COMPENSATION	
• SPLIT-STERLING CYCLE FPA REFRIGERATOR	

Figure 9. TacSat Multi-Purpose MSI Sensor Concept

Multi-Spectral Band Selection

The earth's atmosphere has many highly absorbing molecular species. Their influences establish spectral windows for satellite sensors to detect surface or near-surface phenomena. Waveband options must be chosen considering target phenomenology and these windows.

BAND	WAVELENGTH, μm	MISSION	PURPOSE
1	0.46 - 1.00	VISUAL SURVEILLANCE, INSPECTION, WEATHER	VISUAL MAPS (PANCHROMATIC) DAYLIT CLOUDS
2	0.46 - 0.48	OCEANOGRAPHIC	BLUE - OCEAN DEPTH
3	0.50 - 0.52	OCEANOGRAPHIC	GREEN - DEEP WATER DEPTH
4	3.40 - 3.80	OCEANOGRAPHIC, WEATHER, IR SURVEILLANCE	SEA WAVES & TEMPERATURE, IR MAPPING, NIGHT CLOUDS
5	4.60 - 4.80	OCEANOGRAPHIC	LARGE SHIP DETECTION
6	8.50 - 9.00	IR SURVEILLANCE	LWIR MAPPING, TERRAIN, SHIP DETECTION
7	10.00 - 11.00	WEATHER	CLOUD HEIGHT, IR MAPPING
8	11.00 - 12.00	WEATHER	WATER VAPOR CORRECTION, LAND TEMP., IR MAPPING

Figure 11. MSI Sensor Bands & Uses

Multi-spectral information is highly useful for many TacSat missions. When used to supplement panchromatic data, the non-visible bands add detectability. Camouflage can be detected as can warm objects against a cooler background. Multiple band measurements can be used to compensate data for environmental conditions such as water vapor effects. The bands listed in Figure 11 represent an initial selection based on expected targets and backgrounds of greatest interest and likelihood. The system and sensor's programmable features allow specific uses of band combinations for each mission with data formatted for transmittal to the ground.

Optical Design Selection

All-reflecting optics designs are superior for simultaneous use of different spectral bands. Reflecting optical systems also offer feasible weights in this aperture range. Except for small refractive elements providing image correction in the high-resolution visual band, the design uses all reflecting optics. Detectors for the high resolution panchromatic array are located with the central 1° of the line FOV where geometric aberrations are small. This has the disadvantage of reducing available visual coverage rates, but is necessary to balance distortion and resolution. Using two focal planes with different focal ratios maximizes the combined FOV and resolution.

Physical law dictates that resolution is proportional to the ratio of aperture diameter and spectral wavelength. For apertures feasible on a small satellite, high resolution is attainable only in the visual and near-to-mid-IR bands, from lower orbits. The 45 cm aperture compromise provides good performance (SNR and resolution) for most missions, and construction with reasonable size and weight. For mapping, SNR generally decreases and resolution increases with f/#. **The f/7 scanning-mode optical path is a compromise between SNR and resolution performance.** In staring, high resolution-mode, a higher f/# is desired. **With a large square silicon CCD detector, an f/15 system meets resolution requirements with good FOV and SNR performance.**

For given aperture and resolution, wide-field-of-view sensors tend to be large. The design has a slightly off axis optical path providing a wide, one dimension FOV, with resolution only partially degraded by geometric aberrations. This approach is acceptable for IR bands, where diffraction blur exceeds the geometric blur. An additional square high-resolution focal plane uses the on-axis optical path and operates only in the visual band, providing maximum resolution for designated areas of interest. Operating the highest resolution missions solely in the shorter wavelength visual band keeps the aperture size small. When sunlit, most ground and weather features of interest show visual-band contrast adequate for detection.

The square visible staring array receives the on-axis narrow-FOV beam through a central hole in the primary mirror. The multi-spectral optical path lies just to the side of the staring systems lens elements. Figure 12

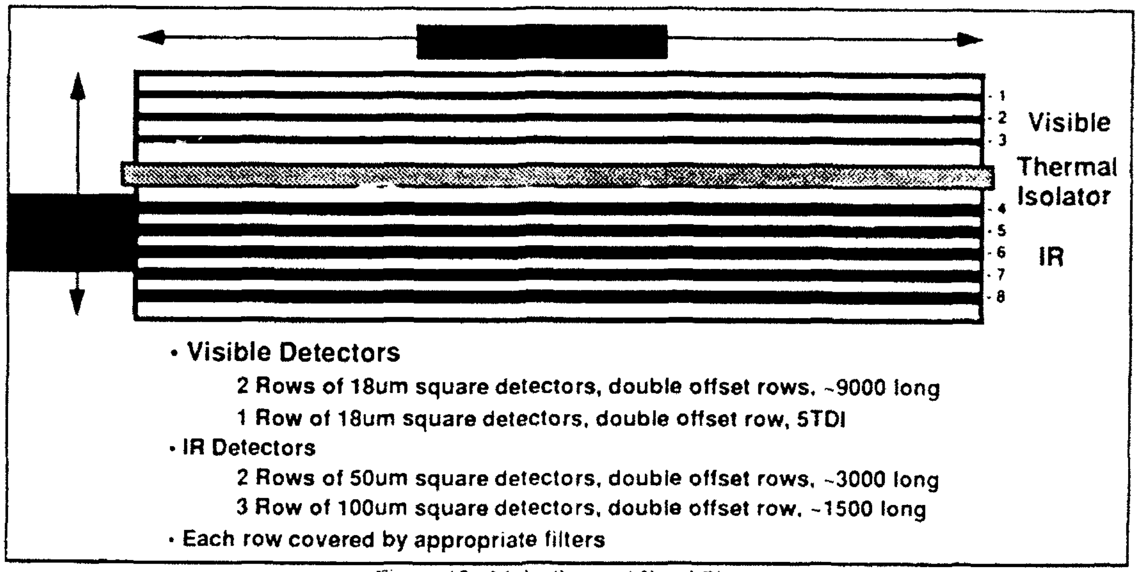


Figure 12. Multi-Spectral Focal Plane

shows the multi-spectral Focal Plane Array (FPA) layout. A thermal isolator divides the FPA into an uncooled visual and cooled (80 °K) IR waveband sections. Filters to define the detected waveband cover the detectors in each pair of staggered-alignment rows, with multi-color data received automatically as the rows sweep across the scene. The FPA length will likely require use and alignment of separate smaller line array sections combined to form the FPA. Signal-to-noise ratio analysis of a scene to be imaged or mapped must account for the observer's desired contrast level. Background noise rather than internal sensor noise limits TacSat mapping observations. This is because the earth is bright in most wavebands, especially the sunlit half. The result is the parabolic relation of contrast to photon count. Since detectable contrast depends on photon count, for background-limited detection the relation between contrast level, required SNR, scan rate, and spectral bandwidth, is:

Contrast = SNR(Scan Rate)^.5 (Sp. Bandwidth)^.5.*
This helps to choose the scan rate, once the expected target and background signatures, and/or search areas are defined. A commandable sensor allows operations tailored to immediate mission requirements.

Sensor operations could involve bi-directional whisk-broom scans covering about a 14°-wide swath parallel to the flight path at up to 45° from nadir. Desired ground resolutions are challenging to achieve over the full FOV with a moderate aperture, especially in the IR bands, and a wide field-of-view with reasonable and weight is needed for these missions. Data downlink rates are high for mapping and surveillance missions. This makes it difficult to process and return data within time constraints. These reflect the combination of wide coverage and medium resolution. If attention is limited to a few areas, slower scan and data rates are possible. Figure 13 shows the sensor block diagram.

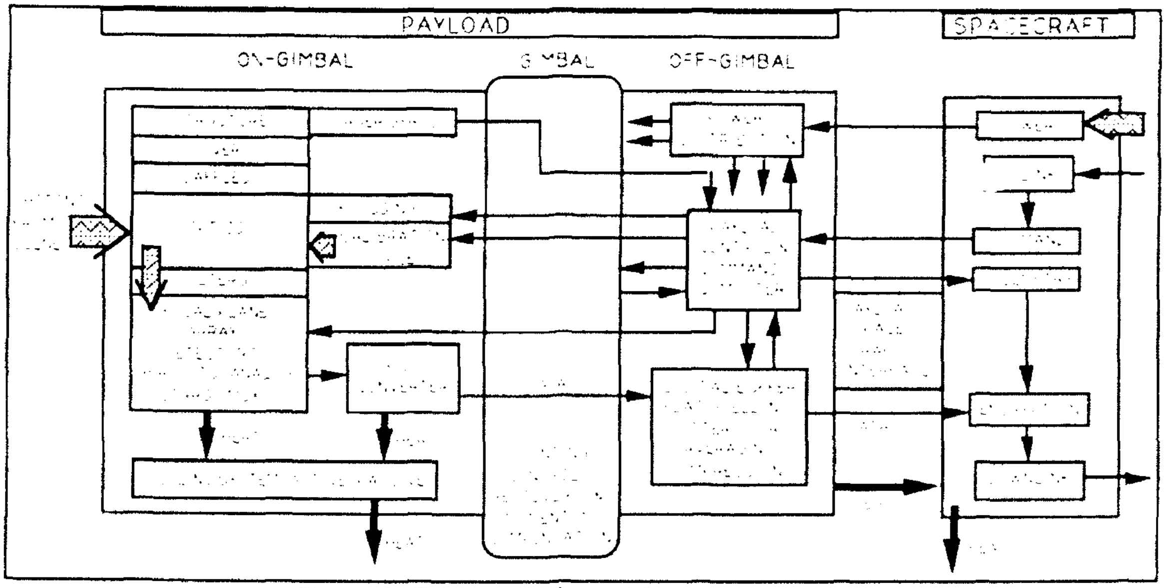


Figure 13. TacSat Multi-Spectral Sensor Block Diagram

Sensor Thermal Design

With cooled IR band FPA detectors, adequate sensor heat rejection is essential. Operation of infrared sensors requires cooling of detectors and other local thermal photon noise sources (optics and structures). Detectors, if HgCdTe, operate with sufficiently low dark current when at 80°K. To achieve such temperatures without large radiators, mechanical coolers are needed. Two small radiators are used, one for the mechanical cooler and on-gimbal electronics, and the other to keep the optics and structures cool. A lightweight, rapid-readiness, cooler with long storage life could be required for TacSat missions. In the desired size range, a mechanical Sterling cycle cooler is a good choice. It operates at a coefficient of operating power of approximately 100 Watts consumed per Watt of cryocooling dissipation.

Since the sensor is designed to operate in the "push broom" or "whisk broom" mode, it is unlikely that a radiator would be continuously shaded. Some alternatives are: periodically moving the primary spacecraft nadir-pointing axis off nadir, fitting a sun shield, or using a sun-synchronous orbit and "whisk broom" mode exclusively. It might also be possible to change operational modes so that radiator sun exposure is less than 35 minutes per orbit. However, the combination of a sun shield and alternating modes could allow shaded operations with minimal mode use interference and allowing inclination choice freedom.

Satellite Design

Figure 14 shows a possible TacSat configuration. The TacSat bus houses all equipment to support the sensor payload and communications equipment. The key to TacSat flexibility and utility is a bus that uses a single Integrated Spacecraft Electronics (ISE). Figure 15, the spacecraft block diagram, shows major vehicle components and interfaces and how the ISE controls and coordinates functions. The ISE processes attitude sensor data and controls orbit and attitude through the reaction wheels and the propulsion system. The ISE also collects and formats spacecraft housekeeping data for storage and/or transmission.

The ISE is the "heart" of the TacSat spacecraft. It is a single electronics package with a modular data bus. A series of mission and application specific cards attach to the data bus. These cards connect the bus systems, and payload if desired, to the data bus. A high speed CPU is also on the data bus, as shown in Figure 16. Locally generated data and ground commands move to the processor. Once in memory, this data is used to generate spacecraft or transmission commands.

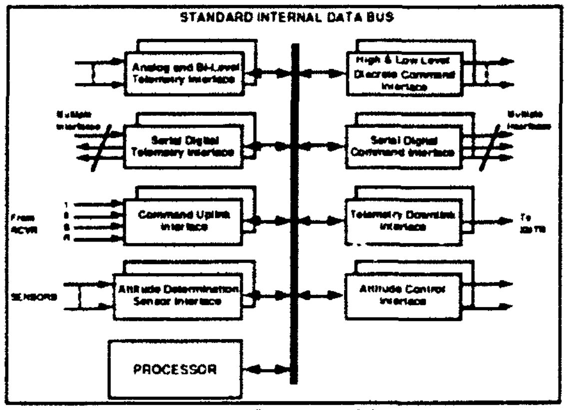


Figure 16. ISE Block Diagram

Electrical Power Subsystem

The electrical power system (EPS) design should be simple and flexible. The system uses either gallium-arsenide-on-germanium or silicon solar cells and nickel-hydrogen batteries. Gallium-arsenide solar panels, while more expensive, offer improved packaging, lower weight and reduce orbital disturbance torques and moments of inertia compared to silicon cell panels. A single pressure vessel (SPV) nickel-hydrogen battery can lower both weight and cost compared to an equivalent individual pressure vessel (IPV) system. These factors combine to reduce propellant mass.

Attitude Control and Propulsion

Three-axis stabilized TacSat attitude control and propulsion subsystem functions provide a stable sensor platform and include: determination of attitude and

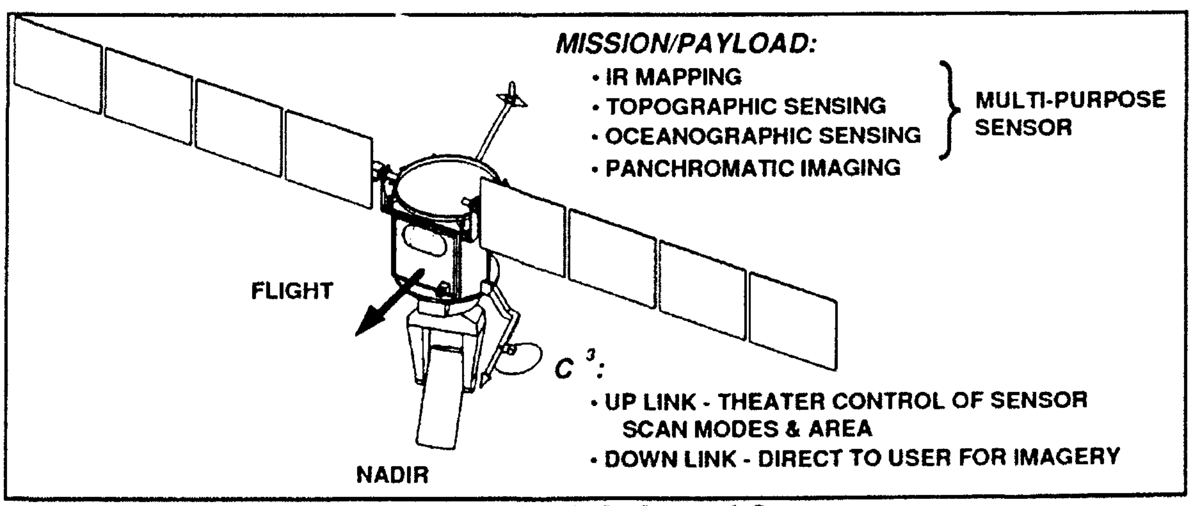


Figure 14. Possible TacSat Spacecraft Concept

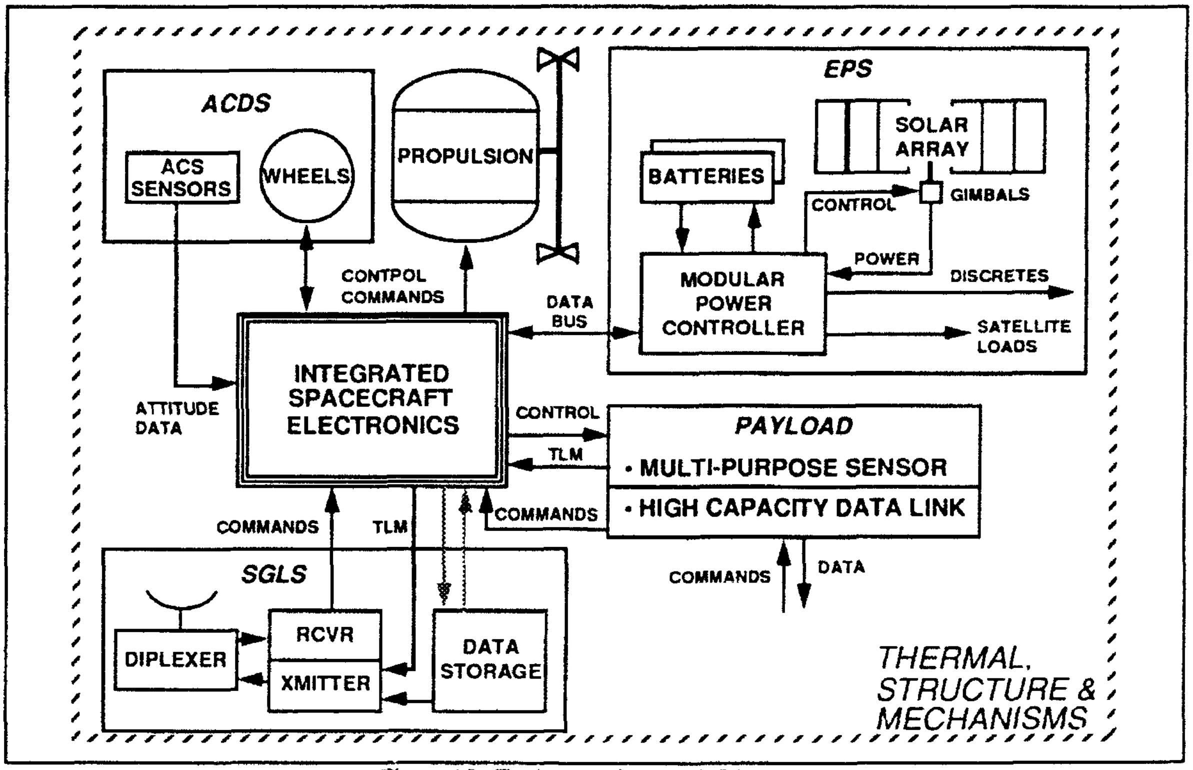


Figure 15. Typical TacSat Block Diagram

position, attitude control and pointing agility, orbit adjust control and actuation, payload torque and momentum compensation, safemode control and de-orbit, if required. The subsystem hardware consists of sensors, actuators and propellant storage and delivery system. All control law, safemode and other software reside in the ISE. Three reaction wheels provide torque actuation, with agility requirements and residual payload torques after momentum compensation driving their size and magnetic torquers for desaturation. Low cost star camera sensors provide the kind of knowledge needed, with a sun sensor for coarse pointing.

The propulsion system consists of a conventional hydrazine mono-propellant tank with propellant management system, lines, filter, valves, instrumentation and thrusters. System pressurization could use either a blow-down or a re-pressurization system. The propulsion system is used to correct 'aunch vehicle injection errors, perform orbit trim maneuvers and de-boost if required.

Structures and Mechanisms

Major bus structure components are: the frame, equipment panels, propulsion module, and interface ring. The modular design allows subsystem mounting while maintaining alignments during test, transportation, and storage. The modular approach also reduces part count and assembly spans. The interface ring provides the structural load path between the bus structure and launch vehicle. Graphite-epoxy composite (Gr/E) was selected over a metallic material, for both primary and secondary structure, after conducting trade studies. Gr/E provides a stable (near zero CTE) structure to meet and maintain attitude control sensor pointing accuracy. The Gr/E structure saves 4.9 kilograms over Al/Li and 7.9 kilograms over Al. The primary structure provides continuity, strength and rigidity between the payload and the interface ring.

Space-Ground Link System (SGLS) and high data-rate antennas stow against the vehicle and deploy on hinges. Spring driven mechanisms deploy both antennas and the solar arrays. Each solar array gimbal is driven by small motors and reducers. Flex harness and slip rings carry electrical power and telemetry across the joints.

Thermal Control

Temperature extremes allowed in other subsystems drive the satellite thermal subsystem. Assuming the general case using a non-sun synchronous orbit, TacSats would not have "hot" and "cold" sides, as the thermal environment varies on the satellite sides depending on inclination and time of year. A TacSat is subject to many eclipse cycles and must be designed to accommodate them. For example, components that operate during the sunlit portion of the orbit would have a significant thermal radiation area to reject both internal and incident heat. Unfortunately, when this surface is in eclipse, heater power must now maintain the item with the larger radiator sized for the full-sun operation case. Albedo and earthshine (planet IR heating) also influence LEO vehicle environments.

TT & C

Figure 17 shows the two major communication system hardware groups. User support is prov.ded by an encrypted two-way common data link (CDL) at X-Band. A small steerable downlink dish can provide about 274 megabit/sec in LEO orbits. The up-link provides ample capacity into an omni antenna at 16 or 200 kbps. This is the primary user payload control link but could support housekeeping, as an SGLS alternate.

Centralized control and maintenance reduce the user's workload. Because the spacecraft would perform normal systems management, housekeeping could be done from a small ground station, collecting telemetry, correcting anomalies, etc. Continued house-keeping for a large constellation requires a larger mission control center. Existing launch command and control resources are a choice for launch and early orbit functions.

SPACECRAFT CONFIGURATION	CAPACITY	USEAGE
CDL X-BAND DOWNLINK • .6 METER STEERABLE DISH • 5-10 WATT TRANSMITTER	DIRECTIONAL COVERAGE AT 274 MEGABITS/SEC WITH A 2 METER MOBILE TERMINAL	HIGH RATE, WIDE BAND IMAGERY DATA LINK
CDL X-BAND UPLINK • OMNI ANTENNA (8)	WELL IN EXCESS OF NEED, 16 & 200 Kbps	PAYLOAD POINTING & MODE CONTROL UP-LINK SGLS HOUSEKEEPING ALTERNATE
SGLS TRANSPONDER • 10 CM SPIRAL • 2 WATT TRANSMITTER	HEMISPHERIC COVERAGE AT 1 MEGABITS/SEC WITH A 6 METER TERMINAL	HOUSEKEEPING THRU AFSCN DATA DOWNLINK S-BAND TERMINALS

Figure 17. Typical TacSat Comm & Data Links

Weight and Power

Figure 18 shows a summary level weight and power budget of a typical TacSat in the small satellite (450 kilogram) class. With the use of advanced composites, high efficiency electric power generation and storage devices, and advanced digital technology, a TacSat could achieve a high payload to total dry mass ratio. The spacecraft dry weight is about 45% payload, (sensor and data link) and the rest is the bus. The electrical power requirement estimate is about 480 watts average. The power system is driven by the sensor duty cycle, eclipse operations and cooling power, etc. The 68 kilograms of propellant supports injection and drag-makeup for 1- 2 years.

System / Subsystem	Mass (KG)	Power (W. Avg)
BUS	168	143
PAYLOAD		
MULTI-PURPOSE SENSOR	112	264
DATA LINK	34	75
SUB-TOTAL	314	482
FUEL	68	
GROSS WEIGHT	382	
CONTINGENCY (20%)	76	
TOTAL LAUNCH WEIGHT	458	

Figure 18. Typical TacSat Weight & Power Budget

TACSAT Implementation

Implementation Plan

TacSat concepts have been discussed for many years. Changes in the international climate now make surveillance TacSats the right capability at the right time. Potential users now have a better understanding, gained in the Persian Gulf, of the value of space based surveillance data. System developers have learned to think of space surveillance from the field commander's perspective. Finally, many TacSat system elements are being developed for other purposes. For these reasons, and budgetary realities, the time is right to begin a phased TacSat development and acquisition program.

Since the TacSat concept centers on directly satisfying user needs, implementation efforts must include continuous user involvement. This involvement could start with two simulations. The first would provide users with sample sensor data of various scene types to assess their utility and value. The second simulation would educate potential users about TacSat command and control strategy, planned sensor tasking and how they would receive, process and exploit the data.

While both simulations could be done in a laboratory environment, a portion of the process should be folded into military exercises. The user feedback will establish quantitative criteria for the TacSat system design based on user needs. Simulated sensor data for such an evaluation program could be generated before the event using processed airborne imagery. Imagery of typical battlefield objects and subsequent computer processing to vary image quality would allow the users to fully assess the utility of various data types and qualities.

Once design criteria are established, the TacSat demonstration and validation program would begin. The program would be initiated with the first TacSat and a few user ground terminals. Activities planned, or now underway, indicate that most of the systems required for a demonstration will be available "off-the-shelf." The major exception is the multi-spectral sensor. However, its development risks are nominal as most of the components are either commercially available or of straightforward design and fabrication.

It is likely that the spacecraft bus will be available as a derivative of the U.S. Air Force's Advanced Technology Standard Satellite Bus (ATSSB) or other program. There should be at least one viable candidate launch vehicle for this Dem-Val phase. The Taurus booster should be available and it is likely that another launch system will emerge to support IRIDIUM program operations and maintenance launches.

Satellite control for the TacSat Dem-Val phase could be done effectively as a factory effort by the developer. Commercially available hardware would be employed for user ground stations. Software would be a combination of existing program and unique code.

A Dem-Val program would allow demonstration of:

- Distributed C^3 Architecture Proof of Concept
- Low Cost Satellite Control
- Multiple User Support
- User Tasking Effectiveness
- Local User Processing & Exploitation

The demonstration program would provide an initial operational capability that could expand, as shown in Figure 19. The infrastructure, once established, could support other payloads to form a mixed, highly capable tactical surveillance system.

A phased, "Dem-Val" TacSat implementation strategy would provide a test-bed to evaluate TacSat system utility, provide multi spectral imagery in many bands covering a variety of scenes, assess the benefits of timely, medium resolution MSI, provide an interim treaty monitoring capability, and familiarize military users with tasking , direct control and data exploitation.

System Component	Dem-Val System →	Operational System
· PAYLOAD	· MULTI-SPECTRAL SENSOR	· REPLICATE DEM-VAL SENSOR, ENHANCE FOCAL PLANE
· SPACECRAFT BUS	· LOW COST, HIGH PERFORMANCE BUS	· REPLICATE BUS
· SATELLITE CONTROL	· CONTRACTOR & USERS	· MILITARIZE FOR ENDURABILITY - INCREASED REDUNDANCY - REDUCED CONTRACTOR SUPPORT
· TASKING/ EXPLOITATION	· 1-2 FIXED STATIONS · 1-2 MOBILE STATIONS	· INCREASE USER TERMINALS · ADD RELAY LINK FOR CONUS ONLY USER SUPPORT

Figure 19. TacSat Operational Transition Plan

Deployment and Launch Options

An operational TacSat system must provide a high level of dependability and timely access to data. The number of satellites available directly drives the system's ability to satisfy these needs. A single satellite can only provide a few opportunities per day to image a given target area. If there are conflicting data requirements from multiple targets in a given geographic area, or if there are specific time windows or other constraints, a single satellite may not provide timely support. In addition, the dependability of a single satellite will always be at risk. Therefore, it is very likely that an operational TacSat system will require, in some way, multiple satellites. This need can be satisfied in one of two ways, and selection will be heavily affected by the launch capabilities that emerge in the next few years.

If a quick reaction launch system becomes available, a flexible approach to a TacSat capability, when and where it is needed, would be practical. Satellites and launch vehicles could be stored for call-up and launch in days when a security emergency requires support. To maintain readiness, a satellite would occasionally be launched under exercise conditions. These would provide a level of continuous capability for non-crisis activities, such as training and treaty monitoring. With this capability, satellites could be launched into situation-specific orbits to increase the available coverage for a given emergency.

The absence of a quick reaction launch system would lead to a different strategy. This scenario would be served by in-place TacSats, forming a permanent constellation. Here, the satellites would be in general purpose orbits covering the earth. A set of near-polar sun-synchronous orbits would be likely. For coverage and dependability, the constellation would consist of two or three satellites. Their phasing would be different but altitude and inclination would be similar. Assuming current boosters, the concept is to launch the entire constellation on a single medium booster. A similar launch every few years allows for planned replacement, constellation enhancement or capability upgrades.

Initiating a Dem-Val process for TacSats does not require an early answer to the question of how an operational system will be deployed and launched. There is a good chance that the IRIDIUM program's O&M requirements will provide the foundation for a quick reaction launch system with adequate capability for TacSat use. If this does not occur, the permanent constellation of multiply launched satellites can satisfy operational needs with existing launch vehicles.

SUMMARY

We now have the opportunity to provide our military forces with a cost effective force multiplier. This opportunity exists because:

1) Military organizations are learning how data from space can be used to enhance the effectiveness of existing and planned force structures.
2) The technologies needed for smaller, very capable surveillance satellites exist, generally off-the-shelf.
3) Commercial space programs and various government developments are providing the economic foundation that will make most key TacSat system elements available at affordable costs.

The single major unfunded development required before launch of the first TacSat is for the sensor. Undertaking that development would allow demonstration of the tactical benefits of multi-spectral imagery and lead to the introduction of growing capabilities for military support from space.

In an unstable global security transition era, flexible, multi-purpose space assets are essential. Multi-spectral imaging offers tactical users enhanced capabilities not currently available. A user tasked, multi-role MSI sensor and spacecraft can form a practical TacSat system. The premise for and the validity of TacSats have been substantiated based on the technical analysis presented. It is the authors' opinion that TacSats should be implemented with a sense of national and allied urgency and necessity.

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Ms. V.M. Kilston	LMSC
Mr. J. Kehret	LMSC
Mr. M. Rhynard	Booz, Allen & Hamilton Inc.

Discussion

Question: Would a reduction in altitude allow the size, mass, and cost of the optical system be reduced?

Reply: Yes, a lower altitude would reduce the optical system weight but at the expense of reduced coverage. Also, the increased drag would increase fuel requirements; thereby, offsetting the optics weight reduction. The likely affect is a push on weight and reduced coverage.

Question: What is approximately the total mass of the space segment under discussion or in your paper?

Reply: The approximate mass of the concept is 458kg including fuel and a contingency of 20%.

TACSAT METEOROLOGICAL PAYLOADS

by

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Abstract

Information derivable from Meteorological satellites are reviewed in terms of their applicability to military use. Potential TACSAT meteorological systems are discussed in terms of GEO and LEO payloads currently in operation.

1. Introduction

Knowledge of changes in weather conditions could provide essential information in numerous military scenarios. Such information has been obtained for many years from both civil and military satellite systems and is now a well established technology with the accuracy of information retrieval and processing continually improving.

Meteorological data can relate to either actual regional variations or to predictions of changes for a given area. Both types of information could be of considerable use to a ground commander. For military operations involving the movement or deployment of men and equipment, frequent and accurate regional weather maps and forecasts would be of great assistance with operational planning. Furthermore, meteorological data would also play a significant role in the tactical and strategic deployment of forces in both defensive and offensive operations.

The present paper aims to explore the applicability of meteorological measurements from dedicated satellite systems to support possible tactical military operations. The principles of atmospheric physics and the theory of meteorological data processing are outside the scope of this paper and are not considered here. Of more importance is the relevance of atmospheric parameters which could be measured through a dedicated tactical meteorological system. These requirements are reviewed in section 2.

To ascertain what information could be retrieved from such a TACSAT payload, section 3 discusses the capabilities of a number of current civil and military meteorological payloads which, it is felt, demonstrate how the requirements of a ground commander can be met. The central discussion will be based on the METEOSAT series of satellites together with its companion satellites which form part of a global monitoring system. Although a full review of meteorological payloads cannot be given here, the payloads which are discussed are considered to be relevant to TACSAT applications and indicate what measurements can be performed.

Satellite systems can take the form of either Geostationary Earth Orbit (GEO) or Low Earth Orbit (LEO), the choice of orbit being dependent on the resolution, ground coverage and data rates as well as the scan characteristics of the satellite payload. A further distinction between GEO and LEO satellites is that the former employs passive sensors whereas the latter can also utilise quite recent developments in active sensing of the atmosphere. Active sensors (such as LIDARS) are not considered in any detail in this paper. However, it should be noted that they are becoming increasingly important and do represent important advances in meteorological systems.

In section 4, the features identified in section 2 and the systems presented in section 3 are brought together and discussed in the context of meteorological TACSAT payloads. Additional needs of TACSAT systems which are not currently met by meteorological payloads are reviewed in terms of extensions to present satellite systems. Finally, an assessment is offered on the usefulness of meteorological payloads in a TACSAT role.

2. Requirements Review

Atmospheric conditions clearly play a major role in military operations and must influence decisions made by a local commander. Parameters of interest include:

i) cloud location;
ii) direction and speed of cloud movements;
iii) degree of cloud cover;
iv) cloud altitude;
v) wind speeds and direction;
vi) rain and snow conditions;
vii) flood warnings;
viii) fog conditions.

Cloud location, cover and altitude are important measurements required for both day and night operations since they limit airborne visibility and ground illumination. Additionally, cloud cover controls ground temperature which in turn impacts on personnel and equipment. Coupled to this is the direction and speed of cloud movement which can be used to provide forecasts of cloud cover.

Wind speeds and directions are also important parameters particularly at a ground level. For example the deployment of smoke screens and firing directions of weapons are dependent on wind velocity. In desert terrains, wind conditions will govern sand-storms.

Rain and snow not only affect visibility but are also be very restrictive for ground operations. Warnings of potential flooding and fog condition would also be of obvious use to a ground commander.

Meteorological measurements of the above parameters are required during both day and night.

In addition to the measurements of the conditions present in a given region, other aspects of meteorology must be considered. These include the rate at which weather information is updated as well as the forecasting period. In order to provide a major advantage, the meteorological data should be updated on a timescale of minutes rather than hours. Another important factor is that the meteorological measurements and forecasting are already both accurate and reliable.

Finally, the format of the data presented to the local commander needs careful consideration to ensure that the essential information is readily accessible and clear. Furthermore, the method and technology available for transmitting / requesting the data must be compatible with the theatre of military operations.

3. Present Meteorological Payloads

3.1 Meteorology - Global Monitoring

The World Meteorology Organisation (WMO) is responsible for providing global coverage of meteorological data. The network is illustrated in Figure 1.

Currently, there are five GEO satellites positioned at various points around the equator at an altitude of approximately 36,000km. In addition to these, there are a number of polar satellites (in a LEO) which operate at an altitude of approximately 850km. Table 1 provides a summary of some of the key features of these meteorology satellites.

The primary role of the geostationary satellites is to provide cloud imagery at regular intervals of approximately 30 minutes. Meteosat, which forms part of the GEO ring is discussed in more detail in the next section.

Apart from their imaging role, the ring of satellites also serve a number of other functions including the transmission and relay of meteorological data from both ground stations and other satellites. They also can be used to relay processed meteorological data between the ground processing centre and the users of data.

The polar satellites are sun-synchronous and each orbit is stepped with respect to the previous such that a full earth coverage is achieved twice per day from each satellite. The main purpose of these satellites is to monitor clouds, surface temperatures and vegetation cover.

3.2 Meteosat Operational Programme

The basic design Meteosat Operational System and its main instrument, the Imaging Radiometer, is over 25 years old. However, because of the success of the Meteosat Operational Programme (MOP), a further model (with possibly a second to follow next year) is currently being built. This is the Meteosat

Transitional Programme (MTP) and bridges to gap to the Meteosat Second Generation (MSG) as discussed in section 3.3. A historical review of the Meteosat Programme can be found in reference 1.

Meteosat is a spin-stabilised satellite whose axis of rotation is aligned with the Earth's North-South Poles. The east west scan is achieved through the rotation of the satellite and the north south scan is accomplished by a scanning mirror.

The key parameters of Meteosat are summarised in table 2.
Meteosat has three key roles. Firstly, it provides images of the Earth at longitude 0^{o} which are then transmitted to a ground processing station. Its second role is to distribute the processed data to user stations and, thirdly, it provides a point of data collection and distribution from other meteorological stations.

The raw data is processed at a ground station and the following information is derived:

- cloud-motion and winds
- sea-surface temperatures
- cloud top height maps
- cloud coverage data
- precipitation indices

The processed images are transmitted, via the Meteosat down-link channel, at a transmission rate of 2400 b/s. The resolution of the processed images is lower than that of the radiometer. Typically, the cloud top height maps are resolved on a grid size of 20km whilst the grid for the other parameters is 200km.

Meteosat comprises three spectral bands, one visible and two IR bands. The visible channel is used to provide high resolution cloud images during daylight. The thermal IR band provides images continuously since it relies on the thermal radiance from the clouds. The IR WV channel provides an atmospheric distribution of humidity in cloud free regions and also provides images of cloud tops.

Two detector units are used, one for the visible channel and one which combines the IR WV and IR. The detectors as such are single element devices with very high (quantum-limited) performance, the layout being illustrated in Figure 2.

3.3 Meteosat Second Generation

MSG is the replacement system for the payload design in the operational and translational programmes. It comprises more waveband channels in the WV and IR and has multiple narrow spectral bands in the visible spectrum. Like MOP and MTP, the MSG satellite system will be spin stabilised, with a spin-rate of 100rpm. Ground resolution is improved, corresponding to 1.4km in the visible and 4.8km in the WV and IR.
A notable feature of the proposed system is that it produces a full Earth image in 12 minutes. The calibration and stabilisation time is 3 minutes and the subsequent repeat cycle is 15 minutes.

3.4 DMSP

The DMSP (Defense Meteorological Satellite Programme) is a US military programme comprising two orbiting satellites. The flight history of the DMSP began in 1965 but remained classified until the mid-70's. The satellite operates in a heliosynchronous orbit of altitude of approximately 800 km and is a 3-axis stabilised platform.

The DMSP satellites contain three payloads. The first of these is the OLS (Operational Linescan System) which provides visible and IR cloud imagery. The second instrument is the SSM/I (Special Sensor Microwave / Imager) which gives information on precipitation, water vapour, snow cover and sea / glacier ice. The third instrument is the SSM/T (Special Sensor Microwave / Temperature) and this provides surface temperatures, vertical moisture and temperature profiles.

4. Discussion

From the few examples described in section 3, it can be stated that present meteorological systems are capable of measuring many of the parameters indicated in the requirements review of section 2.

The concept of the five GEO satellites which provide global coverage is appealing for potential TACSAT systems since continuous monitoring of a ground position can be achieved. Furthermore, the ability to reposition satellite platforms in orbit in order to maximise system sensitivity at a given longitude ensures that the spatial resolution is optimised.

The data processing and re-distribution of meteorological information of current system also lends itself to the TACSAT concept. However, the format of data presentation may need further development to ensure that the information presented is in the most useful form. There may also be a requirement for 'real-time' interrogation of the TACSAT system.

Through the appropriate selection of spectral channels, meteorological data is obtained on a 24 hour basis. By selecting the appropriate spectral bands and their associated spectral widths, the most appropriate information can be obtained. Although present meteorological systems provide much of the data likely to be required by the theatre commander, the emphasis on spectral bands for a TACSAT may be different and a full analysis would be required. Indeed, the system demands may be such that a variable band-pass should be employed through the use of diffraction gratings or tunable interference filters.

For a TACSAT system operating in GEO, it is unlikely that full Earth coverage will be required. A possible concept would be zones of meteorological data centred on the location of interest. The observation or dwell times on a given region would then be controlled through a preset weighting scale or by request from a ground commander.

A possible need from a GEO TACSAT system is to produce higher resolution images than are currently available. If such a need is to be satisfied, the following points will need to be considered:

i) scan mechanism / beam deflection
ii) signal to noise ratio
iii) detector configuration
iv) achievable resolution
v) satellite platform

Firstly, the satellite platform can be either spin or three-axis stabilised. For the former case, the satellite spin gives one scan axis. However, this is generally a fixed value and cannot be optimised for a given application. A three-axis stabilised system has its longitude and latitude scans generated through a scan mirror arrangement. This then provides a more flexible system. Additionally, the dwell time for signal integration can be selectively controlled to optimise signal to noise ratio, image repetition frequency and image size.

The detector configuration is also an option which needs further consideration. For MOP, single element detectors were used and this provided a relatively simple imaging system. Alternatively, a staring array could be used. In this case a boresight pointing system would be required. Although this may be an attractive option since the scan mechanisms are simplified, it may be difficult to obtain an IR array of the required dimensions. For example an area of 1000 km^2 with a resolution of 1 km would require a 1024 x 1024 array.

A different approach may be to use a linear array whose projected image is swept across the area of interest. Large linear arrays can be produced by abutting techniques or by having arrays in physically distinct but in an optically overlapping image plane. Other points would, however, have to be considered before adopting such an approach. These include signal to noise ratios, multiple waveband channels and detector cooling.

The telescope system must be capable of providing the resolution within its diffraction limit. This then sets a requirement on the telescope aperture and angular magnification. For the case of 1 km resolution, an aperture diameter of approximately 500 mm would be required with a magnification of typically 10. Current telescope system designs have shown that these parameters can be readily met.

A major benefit from a TACSAT meteorological capability will be independence from civilian systems and national forecasting services so that in the event of a major crisis during which the civil assets may be switched off for reasons of security, meteorological forecasting is not lost to the Theatre commander.

A viable forecasting system could be supported by 2 or (preferably) 3 satellites which would provide 12 or 8 hour repeat cycles. A payload of a single radiometer operating in the IR would provide imagery at a sufficiently fine spatial resolution to support a sounding capability yielding data for cloud cover, rain cloud temperature, cloud top height (when used in conjunction with a numerical model), and forecasting over several days.

If short term forecasting only was required, a geostationary TACSAT with an IR/optical imager and a limited sounding capability would probably suffice if used in conjunction with a skilled local forecaster.

Both GEO and LEO satellites can provide mechanisms for deriving meteorological data suitable for use by a ground commander. Both have advantages and disadvantages, the importance of which depends on the exact requirements of the TACSAT system. However, it may prove necessary to form a TACSAT from both LEO and GEO payloads. Such an integrated system would provide the maximum flexibility possible for meteorological data extraction.

5. Conclusions

The concept of a meteorological TACSAT system has been addressed through a review of current payloads and military requirements. The use of the GEO systems is seen as being particularly attractive since these can provide continual observation as well as a data transmission link. LEO payloads are seen as a complement to those in GEO and can offer a number of additional benefits. Indeed, for complete atmospheric monitoring and data extraction, both LEO and GEO satellites are necessary.

From review of the possible military requirements, it is considered that present meteorological payloads can provide potentially useful information. Furthermore, current technology can offer improvements which could lead to a viable TACSAT system.

The potential value of a dedicated TACSAT meteorological system appears to hinge on the possible need to replace meteorological data derived from civil sources which may be switched off in time of crisis, and the tactical need for the timely delivery of processed data.

A tactical meteorological system based on 2 or 3 Low Earth Orbiting satellites could provide data adequate for forecasting up to several days ahead, and compensate for the loss of civil data sources, should those be switched off during a crisis. If such assets are not switched off then current GEO systems provide full Earth cloud maps at a refresh rate of typically 30 minutes which is adequate for global monitoring. However, there may be a requirement for military operations in which the provision of a more rapid update over an area much smaller than the global image would be of value. Such a system could be obtained through only a minimal development of current technology and in conjunction with global imagery, would provide sufficient data to enable short term forecasting.

The ultimate assessment of the need for a meteorological system and its subsequent performance characteristics falls to the various Ministries and Departments of Defense. However, it is felt that the technical capability to provide a variety of tactical meteorological systems has been demonstrated.

6. Statement of Responsibilty

Any views expressed are those of the author(s) and do not necessarily represent those of HM Government.

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[1] R.Tessier 'The Meteosat Programme' ESA Bulletin 58, May 1989, pp45-47

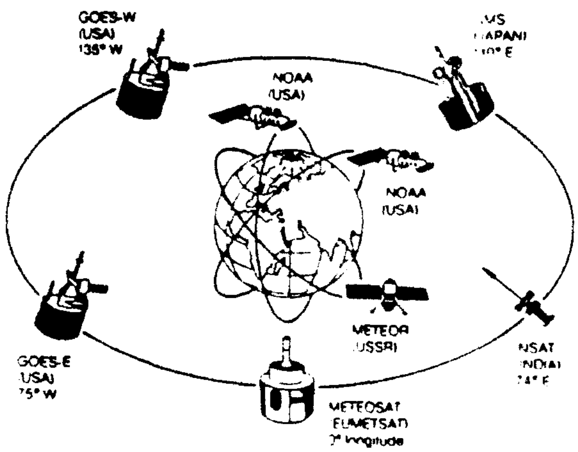

Figure 1 **World Meteorology Organisation**

Satellite	Orbit	Wavebands (um)	Resolution	General
GOES (US)	Geostationary GOES East-75°W GOES West-135° W†	0.55-0.75 3.7-7.3 7.7-14.7	1-8 km 8-14km	Spin Stabilised Full Earth Image in 30 min
GMS (Japan)	Geostationary 140°E	0.5-0.75 10.5-12.5	1.25 km 5 km	Spin Stabilised Full Earth Image in 30 min
INSAT (India)	Geostationary 74°E	0.55-0.75 10.5-12.5	2.75 km 11 km	3-Axis Stabilised Full Earth Image in 30 min Multi-Purpose Communications and Meteorology Satellite
METEOSAT (Europe)	Geostationary 0°	0.5-0.9 5.7-7.1 10.5-12.5	2.5 km 5 km	Spin Stabilised Full Earth Image in 30 min
NOAA (US)	Heliosynchronous 98.7° 850 km 101 min	Various Bands Between 0.58 and 12.5	1.1 km	3-Axis Stabilised 3000 km Swath Width
METEOR (CIS)	Near Polar 950 km 104 min	0.5-0.7 8.0-12.0 10.0-18.0	1-2 km 8 km 30 km	3-Axis Stabilised

† GOES-W failed in orbit and was replaced by moving METEOSAT 3 (spare).

Table 1 **Summary of Global Monitoring Satellites**

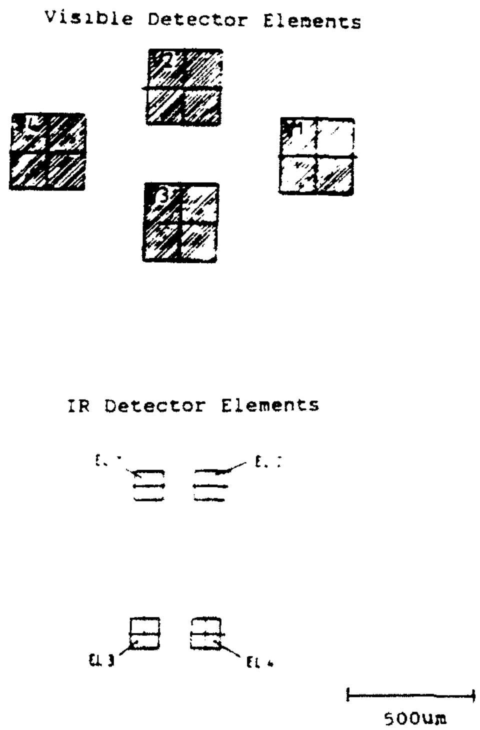

Figure 2 **Meteosat Detector Layout**

Description	Value
Altitude	35,800 km
Diameter / Height	2.1 m / 3.2 m
Mass†	320 kg
Spin-Rate	100 rpm
Wavebands	0.5-0.9 um (visible) 5.7-7.1 um (IR Water Vapour) 10.5-12.5 um (Thermal IR)
Scan-Angle (N-S)	18°
Number of Image Lines (N-S)ƒ	5000 (Visible) 2500 (WV and IR)
Ground Resolution	2.5 km (Visible) 5 km (WV and IR)
Image Formation¶	25 min
Image Repetition§	30 min
Raw Data Rate	333 kb/s

† Includes 39 kg of Hydrazine propellant.
ƒ Complete Earth scan.
¶ Smaller areas can be scanned which will give a more frequent image update.
§ Includes 5 minutes for satellite and radiometer stabilisation.

Table 2 **Meteosat Key Parameters**

Discussion

Question: You put great emphasis on GEO meteorological satellites, but these cannot give full polar coverage. Could you comment on the significance of this incomplete coverage.

Reply: There is an anticipation that if a crisis/theater operation were to occur, then that operation is more likely in lattitudes remote from the poles. The objection voiced to LEO sensors was based on the relatively long revisit times associated with polar LEO satellites. However, near the poles, revisit times are much more frequent and will probably provide short term data.

Panel Discussion

Comment: TACSATS, like all satellites, are useful only as "trucks" which provide data to supported commanders. Where deployment distances and strategic/operational depths are large, as in Desert Storm, spacecraft are useful. In other situations, such as the US intervention in Grenada, spacecraft are useless because of dwell and ... limitations. What we should do is to examine the functions performed (comms, nav, etc.); remember that these functions are as old as warfare and not unique to space; and then find a means to determine when other means of support are more appropriate from spacecraft.

Question: This is a conference about tactical, low cost, lightsat concepts.

Reply: Possibly yes, but the confusion comes from the mere existence of systems providing support to the theater and that dedicated systems can only be deployed if they are low cost, hence lightsat.

Question: There may be problems of jamming TACSATS.

Reply: Yes. Probably more for SAR than optical systems. Nevertheless, means exist to reduce such damaging effects to a radar, in general. It may be worthwhile to develop some tests to better assess how to cope with the jammer threat.

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14. Abstract

This Symposium dealt with small satellites used for tactical (as opposed to strategic) applications that might be of value to NATO. The 11 Sessions covered TacSat concepts and needs, aspects of TacSat applications, TacSat system applications, communication concepts, launch systems, spacecraft bus, advanced technology, radar concepts, electro-optics concepts and two panel discussions. The information generated by the Symposium will be used by Working Group 16 to report on the utility of such satellites and applications to meet future NATO needs.

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